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LOAD-RANGE PERFORMANCE OF TURBINE-PROPELLER ENGINE
IN TRANSONIC SPEED RANGE AND COMPARISON WITH
LOAD-RANGE PERFORMANCE OF TURBOJET ENGINE

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RESEARCH MEMORANDUM

LOAD-RANGE PERFORMANCE OF TURBINE-PROPELLER
ENGINE IN TRANSONIC SPEED RANGE AND COMPARISON
WITH LOAD-RANGE PERFORMANCE OF TURBOJET ENGINE

By Bernard Lubarsky

SUMMARY

In order to investigate the effect on load-range performance of various combinations of compressor pressure ratio and turbine-inlet temperature for the turbine-propeller engine and to provide a means for comparing the load-range performance of the turbine-propeller and turbojet engines in the transonic speed range, an analysis was made of the turbine-propeller engine for flight speeds from 500 to 800 miles per hour, altitudes from 10,000 to 70,000 feet, compressor pressure ratios from 2 to 30, and turbine-inlet temperatures of 1700°⁰, 2000°⁰, and 2300°⁰ R. The assumptions used in this analysis are representative of the best values obtained in practice or in laboratory investigations; for certain critical assumptions where rapid advance in the field may be anticipated, the effect of changes in these assumptions were shown. The variation, with flight conditions and engine operating variables, of the thrust per square foot of engine frontal area, thrust specific engine weight, thrust specific fuel consumption, ultimate range, range with pay load, and comparison of the load-range characteristics of the turbine-propeller and turbojet engines are discussed.

Maximum or near maximum ultimate range could be attained at any of the flight conditions investigated, except 70,000 feet, 700 and 800 miles per hour, with some compressor pressure ratio between 5 and 10 (not necessarily the same for all conditions). At 70,000 feet, 700 and 800 miles per hour, a compressor pressure ratio less than 5 was required. For altitudes up to 50,000 feet, a 2300°⁰ R turbine-inlet temperature gave about a 5 to 10 percent longer ultimate range than a 2000°⁰ R temperature, which in turn gave about a 7 to 17 percent longer ultimate range than a 1700°⁰ R temperature for the range of flight speeds studied. At 70,000 feet, these percentages were increased to about 11 to 16 and 20 to 36, respectively.

For a turbine-inlet temperature of 2000°⁰ R and altitudes up to about 50,000 feet, the load-range performance of the turbine-propeller engine is appreciably better than that of the turbojet engine for

flight speeds up to 600 miles per hour. For a turbine-inlet temperature of 2000° R and an altitude of 70,000 feet, the turbine-propeller engine gave only marginally better or poorer load-range performance than the turbojet engine throughout the range of speeds investigated.

INTRODUCTION

A theoretical analysis to provide insight into the potential aircraft range and the most suitable operating conditions for six types of propulsion system is presented in reference 1. The engine types considered are: compound, turbine propeller, turbojet, turbo-ram jet, ram jet, and rocket. Reference 2 extends the analysis of reference 1 for the turbojet engine and determines, on the basis of load-range performance, the optimum combination of compressor pressure ratio and turbine-inlet temperature in the transonic speed range.

The large improvement in turbine-propeller technology since the preparation of reference 1, which raised the question of the feasibility of flight in the transonic speed range with this engine type, indicated the desirability of also reviewing the performance of the turbine-propeller engine. Accordingly, a detailed investigation was made at the NACA Lewis laboratory of the effect of compressor pressure ratio on the performance of the turbine-propeller engine for turbine-inlet temperatures of 1700° , 2000° , and 2300° R.

The performance of the turbine-propeller engine and the load-range characteristics of aircraft powered by the turbine-propeller engine were calculated for flight speeds from 500 to 800 miles per hour and for altitudes from 10,000 to 70,000 feet; reference 1 covers flight speeds below 500 and above 800 miles per hour. The assumptions used in this analysis are representative of the best values obtained in laboratory investigations; for certain critical assumptions, where rapid advancement in the field may be anticipated, the effect of changes in these assumptions are shown. All of the assumptions in this report are identical to those of reference 2 except the nacelle drag coefficients, which have been changed in the light of more recent information. The load-range performance of the turbine-propeller engine is compared with the load-range performance of the turbojet engine taken from reference 2.

METHODS

A diagram of the turbine-propeller engine assumed for the analysis is shown in figure 1. The performance of the turbine-propeller engine

was calculated for flight altitudes of 10,000, 30,000, 50,000, and 70,000 feet; for flight speeds of 500, 600, 700, and 800 miles per hour; for compressor pressure ratios from 2 to 30; and for turbine-inlet temperatures of 1700°, 2000°, and 2300° R. The compressor was assumed to be of the axial-flow type for pressure ratios up to 10. For pressure ratios greater than 10, the compressor was assumed to be composed of two parts: an axial-flow compressor with a pressure ratio of 10, followed by a centrifugal-flow compressor with whatever pressure ratio was required to achieve the over-all pressure ratio desired. Losses between the compressors were neglected. The engine performance was calculated using the thermodynamic data of references 3, 4, and 5 for the compression, combustion, and expansion processes, respectively. The pressure ratio across the exhaust nozzle was assumed to be equal to the pressure ratio across the engine-inlet diffuser. Reference 6 indicates that this assumption gives near maximum power for the turbine-propeller engine. The pressure drop in the combustion chamber was neglected, as it was found that the pressure drop was small and had a negligible effect on the engine performance.

The following constant quantities were assumed:

Axial-flow-compressor polytropic efficiency (total-to-total)	0.88
Centrifugal-flow-compressor adiabatic efficiency (total-to-total)	.80
Combustion efficiency	.98
Turbine adiabatic efficiency (total-to-total)	.90
Exhaust-nozzle velocity coefficient	.97
Gear efficiency	.95

Two different values of propeller efficiency were assumed; 0.80 for speeds of 500 and 600 miles per hour, and 0.70 for speeds of 700 and 800 miles per hour. The pressure-rise recovery factor (ratio of actual pressure rise to theoretical pressure rise) of the engine-inlet diffuser was assumed to vary with flight Mach number as shown in figure 2. The curve of figure 2 was obtained from a survey of current literature.

The air flow for the turbine-propeller engine was assumed to be 13.0 pounds per second per square foot of engine frontal area for

sea-level zero-ram conditions at the compressor inlet. At other altitudes and flight speeds, the air flow was calculated by assuming that the axial Mach number at the compressor inlet remained constant at the value corresponding to the sea-level static air flow.

$$w = 13.0 \frac{P/P_o}{\sqrt{T/T_o}} \quad (1)$$

(All symbols used in this report are defined in the appendix.)

The weight of the turbine-propeller engine, exclusive of gears and propeller, was assumed to increase with increasing compressor pressure ratio r_c and turbine enthalpy drop, as shown in figure 3. The gears were assumed to weigh 0.1 pound per altitude-cruise-shaft horsepower, and the propeller weight was assumed to vary inversely with the altitude density and with the square of the flight speed about a reference value of 0.2 at 30,000 feet, 600 miles per hour, as shown in the following table:

Flight speed (mph)	Propeller weight, (lb/hp)			
	Altitude, (ft)			
	10,000	30,000	50,000	70,000
500	0.144	0.288	0.720	1.728
600	.100	.200	.500	1.200
700	.073	.147	.367	.882
800	.056	.113	.281	.675

For corresponding compressor pressure ratios, the specific engine weights found by means of these assumptions approximate the specific weights of the lightest of current engines. The gears and the engine were assumed to be submerged in the wing or in the fuselage, except as subsequently noted; where the engine is not submerged, the gears were assumed to have a smaller diameter than the engine.

The following assumptions of airplane characteristics were made: (a) structure- to-gross-weight ratio, 0.4; and (b) fuel-tank- to-fuel-weight ratio, for integral metal tanks, 0.05. The maximum lift-drag ratio that the airplane could attain, not considering wing loading, and so forth, was assumed to vary with flight Mach number as shown in figure 4. The lift coefficient at the maximum lift-drag ratio, which

is necessary for the calculation of the wing loading, was assumed to vary with flight Mach number as shown in figure 5. The curves of figures 4 and 5 were obtained from a survey of the literature in this field. These assumed airplane lift-drag ratios and corresponding lift coefficients were used at all the flight conditions for which they did not result in a wing loading higher than 125 pounds per square foot. If the wing loading resulting from these assumptions was higher than 125 pounds per square foot, the airplane lift-drag ratio and the corresponding lift coefficient were reduced by use of representative lift-drag polars for the different speeds so that the wing loading remained constant at 125 pounds per square foot. This value of wing loading is representative of an aircraft that will perform no violent maneuvers and will need either a large airport or some form of thrust augmentation for take off. The following final values of lift-drag ratio were used:

Altitude (ft)	Lift-drag ratio			
	Flight speed, (mph)			
	500	600	700	800
10,000	16.8	14.0	9.0	7.0
30,000	20.0	17.4	11.7	10.0
50,000	20.0	15.3	11.4	11.0
70,000	20.0	15.3	11.4	11.0

Those lift-drag ratios above the dotted line are wing-loading limited. The engines were assumed to be submerged in the wing or in the fuselage and consequently no nacelle drag existed, except as noted in the following paragraph:

Additional calculations were made at 30,000 feet and 500 miles per hour and at all speeds at 50,000 feet in order to indicate the effect of assuming that: (a) The propeller efficiency was decreased 0.10 and 0.20 and increased 0.10 from the values previously listed; (b) the weight of the propeller and gears was increased 50 percent above the values previously listed; (c) the air flow per square foot of engine frontal area at sea-level zero-ram conditions was 6.5 pounds per second instead of 13.0 pounds per second; and (d) the engines were placed in nacelles instead of being submerged. In the calculations for assumption (d), the assumed values of nacelle drag were:

Altitude, ft	30,000	50,000	50,000	50,000	50,000
Flight speed, mph	500	500	600	700	800
Nacelle drag coefficient	0.035	0.035	0.040	0.19	0.24

These values for the nacelle drag coefficient have been reduced from their proper values in order to compensate for the reduction in fuselage drag due to the possible reduction in fuselage size permitted by the removal of the engines to nacelles.

In order to determine the effect of each engine variable on the load-range performance of the turbine-propeller engine, charts similar to those of references 1 and 2 are used. The dimensionless ratio of disposable load to gross weight W_d/W_g is plotted against the initial fuel rate in pounds per mile per ton of gross weight W_f'/W_g . (It should be noted that the units of the gross weight W_g are pounds in all the equations in this section. In the charts, however, where the ratios W_d/W_g and W_f'/W_g are plotted against each other in order to compare the performance of various turbine-propeller engines, the gross weight W_g has units of pounds in the ratio W_d/W_g , but has units of tons in the ratio W_f'/W_g .) On a plot of this type, straight lines through the origin are lines of constant KR , where K is the ratio of the average to the initial fuel rate and R is the range. The relation for KR is

$$KR = \frac{\frac{W_f}{W_g}}{1.05 \frac{W_f'}{W_g}} \quad (2)$$

From equation (2), it is seen that KR for any point in the charts will be equal to the slope of the line joining that point to the origin divided by 1.05 (the ratio of the weight of the fuel and the fuel tanks to the weight of the fuel alone). As in references 1 and 2, the value of K is calculated by the following equation, which assumes a Breguet type flight plan: (See fig. 6.)

$$K = \frac{\frac{1}{1.05} \frac{W_f}{W_g}}{-\log_e \left(1 - \frac{1}{1.05} \frac{W_f}{W_g} \right)} \quad (3)$$

where

$$W_f = W_d - W_c$$

$$W_d = W_g - W_s - W_e$$

$$W_g = F \frac{L}{D} \quad (\text{for submerged installations})$$

$$W_g = F \frac{L}{D} \left(1 - \frac{D_n}{F}\right) \quad (\text{for nacelle installations})$$

$$W_f' = \frac{fF}{V}$$

The ultimate range is found by setting the pay load W_c equal to zero, for which case

$$\frac{W_f}{W_g} = \frac{W_d}{W_g} = 1 - \frac{W_s}{W_g} - \frac{W_e}{F \frac{L}{D}} \quad (4)$$

or

$$\frac{W_f}{W_g} = \frac{W_d}{W_g} = 1 - \frac{W_s}{W_g} - \frac{W_e}{F \frac{L}{D} \left(1 - \frac{D_n}{F}\right)} \quad (4a)$$

and

$$\frac{W_f'}{W_g} = \frac{f}{V \frac{L}{D}} \quad (5)$$

or

$$\frac{W_f'}{W_g} = \frac{f}{V \frac{L}{D} \left(1 - \frac{D_n}{F}\right)} \quad (5a)$$

where equations (4) and (5) are for the case of a submerged-engine installation and equations (4a) and (5a) are for the case of the engine mounted in nacelles.

RESULTS AND DISCUSSION

Performance of Turbine-Propeller Engine

The various components of engine performance are discussed in the following paragraphs:

Thrust. - The variation of net thrust per square foot of engine frontal area with compressor pressure ratio for a turbine-propeller engine at altitudes of 10,000, 30,000, 50,000, and 70,000 feet, at flight speeds of 500, 600, 700, and 800 miles per hour, and for turbine-inlet temperatures of 1700°R, 2000°R, and 2300°R is shown in figure 7.

For simplicity and convenience, the thrust curves and the curves that follow are not always extended over the entire range of compressor pressure ratios investigated. All the curves of figure 7 show peaking of the thrust with varying compressor pressure ratio, which is characteristic of a turbine-propeller engine with constant turbine-inlet temperature. The maximum thrust varies from about 1225 pounds per square foot of engine frontal area at 10,000 feet altitude, 500 miles per hour, and a turbine-inlet temperature of 2300°R (fig. 7(a)) to about 52 pounds per square foot of engine frontal area at 70,000 feet, 700 miles per hour, and a turbine-inlet temperature of 1700°R (fig. 7(d)). The maximum thrust decreases with increasing altitude and increases with increasing turbine-inlet temperature. The maximum thrust decreases with increasing flight speed to about 700 miles per hour and then increases with increasing flight speed. The compressor pressure ratio for maximum thrust varies with altitude, flight speed, and turbine-inlet temperature, but pressure ratios from 5 to 7 give maximum or near maximum values for the thrust at all conditions investigated.

Specific weight. - The variation of thrust specific engine weight with compressor pressure ratio is plotted in figure 8 for the same range of conditions given for figure 7. The weights of the propeller and the gears are included in the engine weights used in plotting figure 8. The specific engine weight increases with increasing compressor pressure ratio throughout the range of pressure ratios investigated. No minimum occurs with varying compressor pressure ratio because the engine weight decreases more rapidly than the thrust as the pressure ratio decreases from the value necessary for maximum thrust. Specific weight increases with increasing altitude and decreases with increasing turbine-inlet temperature.

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Specific fuel consumption. - The variation of thrust specific fuel consumption with compressor pressure ratio is shown in figure 9 for the same range of conditions given for figures 7 and 8. The specific-fuel-consumption curves have a minimum point with respect to varying pressure ratio, which again is characteristic of a turbine-propeller engine with a constant turbine-inlet temperature. The minimum specific fuel consumption decreases with increasing altitude up to about 35,000 feet, after which it remains constant as the altitude continues to increase. Minimum specific fuel consumption increases with increasing flight speed but decreases with increasing turbine-inlet temperature, varying from about 0.55 at 50,000 and 70,000 feet, 500 miles per hour, and a turbine-inlet temperature of 2300° R (figs. 9(c) and 9(d)) to about 1.17 at 10,000 feet, 800 miles per hour, and a turbine-inlet temperature of 1700° R (fig. 9(a)). The compressor pressure ratio at which the minimum specific fuel consumption occurs decreases with increasing flight speed and increases with increasing turbine-inlet temperature, varying from about 6 at 10,000 feet, 800 miles per hour, and a turbine-inlet temperature of 1700° R (fig. 9(a)) to 28 at 50,000 and 70,000 feet, 500 miles per hour, and a turbine-inlet temperature of 2300° R (figs. 9(c) and 9(d)).

Load-Range Characteristics of Turbine-Propeller Engine

In order to compare the load-range performance of the various turbine-propeller engines considered, charts of the type described in the section METHODS are used. The load-range performance of the turbine-propeller engine is shown in figure 10 for the same conditions given for the engine-performance curves.

If the flight speed and the lift-drag ratio are constant, as they are in each individual plot of figure 10, the ratio W_d/W_g depends only on the specific engine weight and the ratio W_f'/W_g depends only on the specific fuel consumption. (W_d/W_g decreases as the specific engine weight increases and W_f'/W_g increases with specific fuel consumption.) The variation of these ratios with the compressor pressure ratio and the turbine-inlet temperature over the range of conditions investigated therefore follows directly from the variation of the specific weight and the specific fuel consumption (figs. 8 and 9, respectively). The ultimate range and the range with pay load are determined by the values of the ratios W_d/W_g and W_f'/W_g , as described in the section METHODS. For any point on the curves of figure 10, KR is proportional to the slope

of the line joining that point to the origin of the coordinate system (0,0). Three lines of constant KR are shown on each plot of figure 10 for convenience in estimating range. A more complete discussion of this type of chart is presented in reference 1.

Ultimate range. - In order to show more conveniently the effect of the different variables on the ultimate range, the following tabulation of the maximum ultimate range and the compressor pressure ratio at which this maximum occurs for all flight conditions and turbine-inlet temperatures investigated, using figures 6 and 10, is presented:

Altitude (ft)	Turbine- inlet tempera- ture (°R)	Flight speed, (mph)							
		500		600		700		800	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
10,000	1700	7800	8	6500	7	3800	6	2950	5
	2000	9100	10	7600	9	4100	8	3150	7
	2300	9700	15	8100	13	4400	11	3400	9
30,000	1700	10,200	10	9000	9	5000	7	4500	6
	2000	11,400	13	9900	11	5500	9	4800	8
	2300	12,100	18	10,500	15	6000	12	5150	10
50,000	1700	8000	8	5600	7	3300	5	3550	4
	2000	9100	10	6400	9	3700	6	3950	5
	2300	9600	13	7000	11	4000	8	4250	6
70,000	1700	3750	4	2050	3	1000	2	1400	< 2
	2000	4500	5.5	2800	4	1300	2.5	1700	2
	2300	5100	7	3250	5	1450	3.5	1900	2.5

The maximum ultimate range increases as the altitude is increased to about 30,000 feet, and then decreases as the altitude is further increased. Because of the decrease in lift-drag ratio, the ultimate range decreased with increasing flight speed in the transonic region except where the lift-drag ratio remains nearly constant (700 to 800 mph at 50,000 and 70,000 ft); here the ultimate range increases with flight speed. At all flight conditions considered, the ultimate range increases with increasing turbine-inlet temperature, within the range of temperatures investigated. For altitudes

up to 50,000 feet, a turbine-inlet temperature of 2000° R gives about a 7 to 17 percent longer range than a turbine-inlet temperature of 1700° R; a turbine-inlet temperature of 2300° R gives about a 5 to 10 percent longer range than a turbine-inlet temperature of 2000° R. At 70,000 feet, a turbine-inlet temperature of 2300° R gives about an 11 to 16 percent longer range than a 2000° R turbine-inlet temperature, which in turn gives a 20 to 36 percent longer range than a 1700° R turbine-inlet temperature.

The compressor pressure ratio for maximum ultimate range increases as the altitude is increased to 30,000 feet and then decreases as the altitude continues to increase. At all altitudes, the pressure ratio for maximum ultimate range decreases with increasing flight speed and increases with increasing turbine-inlet temperature.

As may be seen from figure 10, the ultimate range falls off slowly as the pressure ratio is varied in either direction from the value necessary for maximum ultimate range; that is, there exists at each flight condition a range of compressor pressure ratios that will give close to optimum performance on the basis of ultimate range. This range of pressure ratios is wider at low flight speeds than at high flight speeds with the same variation in performance. Some latitude in the selection of design compressor pressure ratios for a given application exists because of this band of pressure ratios giving close to optimum ultimate-range performance. Some compressor pressure ratio between 5 and 10 (not necessarily the same for all conditions) will give near optimum or optimum ultimate range at all of the conditions investigated except 70,000 feet, 700 and 800 miles per hour, where a pressure ratio less than 5 is required (fig 10(d)).

Ranges less than ultimate. - Another measure of the load-range performance of the turbine-propeller engine is the range with a given pay load. Figure 11 shows, for a turbine-propeller engine operating at 30,000 feet and 500 miles per hour with turbine-inlet temperatures of 1700°, 2000°, and 2300° R, the variation of range with compressor pressure ratio for values of pay-load- to-gross-weight ratio W_c/W_g of 0 (ultimate range), 0.2, and 0.4.

For the flight conditions of figure 11, the maximum range and the corresponding compressor pressure ratio both decrease as the pay load increases. Although it is not shown, this decrease also occurs for all other flight conditions. As in the case of the ultimate range, there exists for each pay load at each flight condition a range of compressor pressure ratios that will give close to the maximum range.

Although not shown, curves were drawn of the variation with compressor pressure ratio of the range of the turbine-propeller engine with $W_c/W_g = 0.2$ for the same range of conditions given for the engine-performance curves. The following tabulation of the maximum range and the compressor pressure ratio at which the maximum range occurs was taken from these curves:

Altitude (ft)	Turbine- inlet tempera- ture (°R)	Flight speed, (mph)							
		500		600		700		800	
		Maximum range (miles)	r_c	Maximum range (miles)	r_c	Maximum range (miles)	r_c	Maximum range (miles)	r_c
10,000	1700	4300	7	3600	6	2000	5	1500	4
	2000	5000	9	4100	8	2150	7	1600	6
	2300	5500	13	4400	12	2300	10	1750	8
30,000	1700	5500	8	4800	7	2550	6	2250	5
	2000	6100	11	5300	9	2800	8	2450	7
	2300	6500	15	5600	13	3050	11	2650	9
50,000	1700	3800	7	2600	6	1400	4	1600	3
	2000	4450	8.5	3000	7.5	1600	5	1800	4
	2300	4800	11	3300	9	1800	6	1950	5
70,000	1700	900	3	400	2	----	--	100	< 2
	2000	1350	4.5	700	3	100	< 2	350	< 2
	2300	1700	6	850	4	250	< 2	550	< 2

A comparison of this table with the table for maximum ultimate range shows that the maximum range with a pay-load- to-gross-weight ratio of 0.2 and the pressure ratio at which the maximum occurs follow the same trends with altitude, flight speed, and turbine-inlet temperature as for the case of maximum ultimate range. Although not shown, this trend is true for all pay-load- to-gross-weight ratios that do not result in very short ranges (under 1000 miles).

Comparison of Load-Range Performance of Turbine-Propeller and Turbojet Engines

In order to compare the load-range performance of the turbine-propeller and turbojet engines, values for the load-range performance

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of the turbojet engine were taken from reference 2. The assumptions used in reference 2 to arrive at the values for the load-range performance are identical to those listed in the section METHODS. (It is important to note that a theoretical load-range comparison of two different engine types cannot, by its very nature, be precise and any such comparison can therefore show only trends and large differences in performance; where the theoretical comparison shows only small differences between the two engine types, no conclusions can be drawn.) Figure 12 shows the variation with flight speed of the ultimate range and the range with a pay-load- to-gross-weight ratio of 0.2 of the turbine-propeller and turbojet engines for altitudes of 10,000, 30,000, 50,000, and 70,000 feet, with a turbine-inlet temperature of 2000° R and with the compressor pressure ratio that gives the longest range in each particular case. The load-range performance of the turbine-propeller engine has its greatest margin of superiority with respect to the load-range performance of the turbojet engine at the lowest speed investigated (500 mph) at 30,000 feet and with zero pay load, where the range of the turbine-propeller engine is 48 percent longer than the range of the turbojet. The margin of superiority in load-range performance of the turbine-propeller engine is reduced as the flight speed is increased above 500 miles per hour, as the altitude is decreased or increased from about 30,000 feet, and as the pay load is increased above zero. Although not shown, increasing the turbine-inlet temperature within the range investigated improves the load-range performance of the turbine-propeller engine relative to the load-range performance of the turbojet engine. For a turbine-inlet temperature of 2000° R and altitudes up to about 50,000 feet, the turbine-propeller engine gives appreciably better load-range performance than the turbojet up to speeds of 600 miles per hour. For a turbine-inlet temperature of 2000° R and an altitude of 70,000 feet, the turbine-propeller engine gives only marginally better or poorer load-range performance throughout the range of speeds investigated.

Effect of Changes in Assumptions

The effect on load-range performance of a change in some of the assumptions listed in the section METHODS is discussed in the following paragraphs. One assumption is changed in each section. The effects of a change in some assumptions on the optimum combination, on the basis of load-range, of compressor pressure ratio and turbine-inlet temperature of the turbine-propeller engine are evaluated at 30,000 feet, 500 miles per hour and 50,000 feet, 800 miles per hour. The effects of a change in these same assumptions on the comparative load-range performance of the turbine-propeller and turbojet engines are evaluated from 500 to 800 miles per hour at 50,000 feet.

Propeller efficiency. - The propeller efficiency was previously assumed equal to 0.80 at 500 and 600 miles per hour and 0.70 at 700 and 800 miles per hour. The following table shows the effect on the load-range performance of the turbine-propeller engine of assuming that the propeller efficiency is decreased 0.10 and 0.20 and increased 0.10 from the values previously assumed:

Flight conditions	Turbine-inlet-temperature (°R)	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
		$\eta_p = 0.80$		$\eta_p = 0.70$		$\eta_p = 0.60$		$\eta_p = 0.90$	
30,000 ft, 500 mph	1700	10,200	10	8900	9.5	7600	9	11,500	10.5
	2000	11,400	13	9900	12	8400	11	12,700	14
	2300	12,100	18	10,500	17	8900	16	13,700	19
		$\eta_p = 0.70$		$\eta_p = 0.60$		$\eta_p = 0.50$		$\eta_p = 0.80$	
50,000 ft, 800 mph	1700	3550	4	3100	3.5	2750	3	4100	4.5
	2000	3950	5	3400	4.5	2900	4	4550	5.5
	2300	4250	6	3650	5.5	3050	5	4900	6.5

Decreasing the propeller efficiency by 0.10 and 0.20 resulted in decreases in ultimate range of about 13 to 14 and 23 to 28 percent, respectively, and increasing the propeller efficiency by 0.10 resulted in increases in ultimate range of 11 to 15 percent at the flight conditions given in the table. The compressor pressure ratio for maximum ultimate range decreased slightly as the propeller efficiency decreased and increased slightly as the propeller efficiency increased; the variation of ultimate range with turbine-inlet temperature remained about the same.

The effect on the relative ultimate range of the turbine-propeller and turbojet engines of decreasing the propeller efficiency 0.10 and 0.20 is shown in figure 13(a) for an altitude of 50,000 feet, a turbine-inlet temperature of 2000° R, and the compressor pressure ratio that gives the longest range in each particular case. (In fig. 13 and in the following paragraphs, the work "basic" indicates the use of the assumptions of the previous sections.) Decreasing the propeller efficiency 0.10 caused the region where the turbine-propeller engine gives appreciably longer ultimate range than the turbojet engine to be reduced to speeds close to 500 miles per hour. Decreasing the

propeller efficiency 0.20 resulted in the turbine-propeller engine giving, at best, marginally longer ultimate range than the turbojet and this only between 500 and 505 miles per hour.

Propeller and gear weight. - The propeller and gear weights previously assumed are listed in the section METHODS. The effect on the load-range performance of the turbine-propeller engine of assuming that the weights of the propeller and the gears are increased 50 percent above those previously assumed is shown in the following table:

Flight conditions	Turbine-inlet temperature (°R)	Basic propeller and gear weights		Propeller and gear weights increased 50 percent	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	10,200	10	10,000	10
	2000	11,400	13	11,100	13
	2300	12,100	18	11,900	18
50,000 ft, 800 mph	1700	3550	4	3200	3.5
	2000	3950	5	3550	4.5
	2300	4250	6	3800	5.5

Increasing the propeller and gear weight 50 percent above the value used at each particular design point resulted in decreases in maximum ultimate range of about 2 to 3 percent and about 10 percent at 30,000 feet, 500 miles per hour and 50,000 feet, 800 miles per hour, respectively. The compressor pressure ratio for maximum ultimate range remained about the same at 30,000 feet, 500 miles per hour, but decreased slightly at 50,000 feet, 800 miles per hour; the variation of ultimate range with turbine-inlet temperature remained about the same.

The effect on the relative ultimate range of the turbine-propeller and the turbojet engines of increasing the propeller and gear weight 50 percent is shown in figure 13(b) at an altitude of 50,000 feet, with a turbine-inlet temperature of 2000° R, and with the compressor pressure ratio that gives the longest range in each particular case. This increase resulted in a decrease of about 10 to 15 percent in the ultimate range of the turbine-propeller engine as compared with that of the turbojet engine.

Air flow. - The air flow was previously assumed equal to 13.0 pounds per second per square foot of engine frontal area at sea-level zero-ram conditions at the compressor inlet. The following table shows the effect on the load-range performance of the turbine-propeller engine of assuming that the air flow is reduced 50 percent:

Flight conditions	Turbine-inlet temperature (°R)	Basic air flow		Air flow reduced 50 percent	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	10,200	10	8900	9
	2000	11,400	13	10,100	12
	2300	12,100	18	10,800	16
50,000 ft, 800 mph	1700	3550	4	2300	3
	2000	3950	5	2600	4
	2300	4250	6	2950	5

Decreasing the air flow 50 percent resulted in decreases in maximum ultimate range of about 11 to 13 percent and about 31 to 35 percent at 30,000 feet, 500 miles per hour, and 50,000 feet, 800 miles per hour, respectively. The compressor pressure ratio for maximum ultimate range decreased slightly as the air flow decreased; the variation of ultimate range with turbine-inlet temperature remained about the same.

The effect on the relative ultimate range of the turbine-propeller and turbojet engines of decreasing the air flow of the turbine-propeller engine 50 percent is shown in figure 13(c) at an altitude of 50,000 feet, with a turbine-inlet temperature of 2000° R, and with the compressor pressure ratio that gives the longest range in each particular case. Decreasing the air flow 50 percent resulted in the turbine-propeller engine giving, at best, marginally longer ultimate range than the turbojet engine and this only from 500 to 535 miles per hour.

Nacelle installation. - The following table shows the effect on load-range performance of installing the turbine-propeller engine in a nacelle instead of submerging it in the wing, as previously assumed.

Flight conditions	Turbine-inlet temperature (°R)	Submerged installation		Nacelle installation	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	10,200	10	9900	10
	2000	11,400	13	11,100	13
	2300	12,100	18	11,900	18
50,000 ft, 800 mph	1700	3550	4	1250	2.5
	2000	3950	5	2100	4
	2300	4250	6	2750	5

Installing the turbine-propeller engine in a nacelle instead of submerging it in the wing resulted in a decrease in ultimate range of about 2 to 3 percent at 30,000 feet, 500 miles per hour and about 35 to 65 percent at 50,000 feet, 800 miles per hour; the reduction in range decreased as the turbine-inlet temperature increased. The compressor pressure ratio for maximum ultimate range remained about the same at 30,000 feet, 500 miles per hour, but decreased at 50,000 feet, 800 miles per hour; the increase in ultimate range with turbine-inlet temperature that exists for the case of the submerged installation becomes more marked for the case of the nacelle installation.

The effect on the relative ultimate range of the turbine-propeller and turbojet engines of installing the engines in nacelles instead of submerging them is shown in figure 13(d) for an altitude of 50,000 feet, a turbine-inlet temperature of 2000° R, and the compressor pressure ratio that gives the longest range in each particular case. The nacelle drag coefficient used in calculating the ultimate range of the turbojet engine for figure 13(d) is not the same as that used in reference 2, but has been changed, in the light of more recent information, to the values listed in the section METHODS. Installing the turbine-propeller engine in a nacelle resulted in a decrease of about 5 to 30 percent in the ultimate range of the engine as compared with that of the submerged turbojet engine. The relative ultimate range of the turbine-propeller and turbojet engines, when both engines are installed in nacelles, is about the same as the relative ultimate range when both engines are submerged.

Cumulative changes in assumptions. - An indication of the cumulative effect on maximum ultimate range of variations in all four assumptions simultaneously is shown in the following table for the turbine-propeller engine operating at a turbine-inlet temperature of 2000° R, optimum compressor pressure ratio, and with zero pay load:

Change in assumption	30,000 ft; 500 mph		50,000 ft; 800 mph	
	Maximum ultimate range (miles)	Decrease in range (percent)	Maximum ultimate range (miles)	Decrease in range (percent)
Basic turbine propeller	11,400	0.0	3950	0.0
Propeller efficiency reduced 0.10	9,900	13.2	3400	13.9
Weight of propeller and gears increased 50 percent	11,100	2.6	3550	10.1
Air flow reduced 25 percent	10,700	6.1	2850	27.8
Engine installed in nacelle	11,100	<u>2.6</u> 24.5	2100	<u>46.8</u> 98.6
All four preceding changes in assumption simultaneously	8,800	22.8	600	84.8

The table shows that adding the separate effects of various changes in assumptions gives somewhat greater reduction in range than the actual reduction in range due to a simultaneous change of these same assumptions.

SUMMARY OF RESULTS

The results of an analysis of the engine performance and load-range characteristics of the turbine-propeller engine and a comparison of the load-range characteristics of the turbine-propeller and turbojet engines for flight speeds from 500 to 800 miles per hour, flight altitudes from 10,000 to 70,000 feet, turbine-inlet temperatures of 1700°, 2000°, and 2300° R, and compressor pressure ratios from 2 to 30 may be summarized as follows:

1. Although the optimum compressor pressure ratio varied, some compressor pressure ratio between 5 and 10 (not necessarily the same for all conditions) gave near optimum or optimum ultimate range at all conditions investigated except at 70,000 feet and 700 and 800 miles per hour, where a pressure ratio less than 5 was required.

2. For altitudes up to 50,000 feet, a turbine-inlet temperature of 2300° R gave about a 5 to 10 percent longer range than a temperature of 2000° R, which in turn gave about a 7 to 17 percent longer range than a 1700° R temperature for the flight speeds studied. At 70,000 feet, these percentages were increased to about 11 to 16 and 20 to 36, respectively.

3. Of the altitudes investigated, the operating altitude that gave the longest ultimate range at the flight speeds investigated was about 30,000 feet.

4. The ultimate range decreased with increasing flight speed in the transonic region except where the lift-drag ratio remained nearly constant (700 to 800 mph at 50,000 and 70,000 ft); here the ultimate range increased with flight speed.

5. Except for very short ranges, the range of an aircraft carrying a given pay load followed the same trends with altitude, flight speed, and turbine-inlet temperature as did the ultimate range. As the pay load increased, the maximum range and the compressor pressure ratio at which it occurred both decreased.

6. For a turbine-inlet temperature of 2000° R, the range of an aircraft powered by a turbine-propeller engine had its greatest margin of superiority with respect to the range of a turbojet-powered aircraft at the lowest speed investigated (500 mph), at 30,000 feet, and with zero pay load, where the range of the turbine-propeller powered aircraft was 48 percent longer than for the aircraft with a turbojet engine. The margin of superiority in load-range performance of the turbine-propeller engine was reduced as the flight speed was increased, as the altitude was decreased or increased from about 30,000 feet, and as the pay load was increased above zero. Increasing the turbine-inlet temperature improved the load-range performance of the turbine-propeller engine relative to that of the turbojet engine.

7. For a turbine-inlet temperature of 2000° R and altitudes up to about 50,000 feet, the load-range performance of the turbine-propeller engine was appreciably better than that of the turbojet engines for flight speeds up to 600 miles per hour. For a turbine-inlet temperature

of 2000° R and an altitude of 70,000 feet, the turbine-propeller engine gave only marginally better or poorer load-range performance than the turbojet engine throughout the range of speeds investigated.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio.

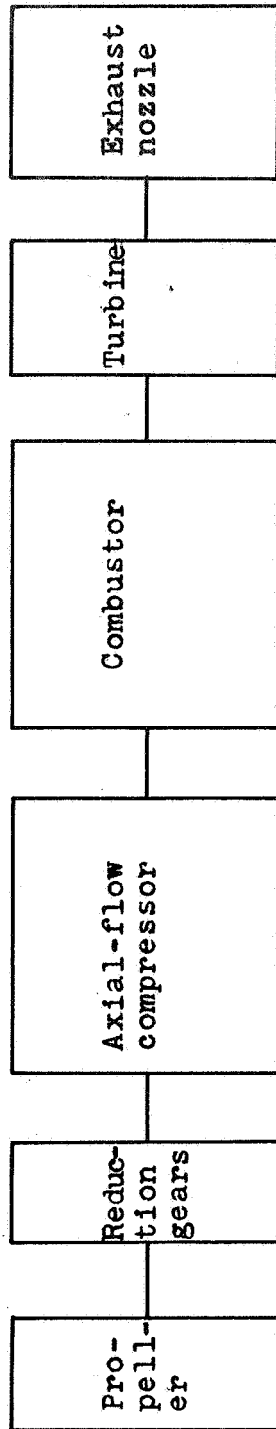
APPENDIX - SYMBOLS

The following symbols are used throughout this report:

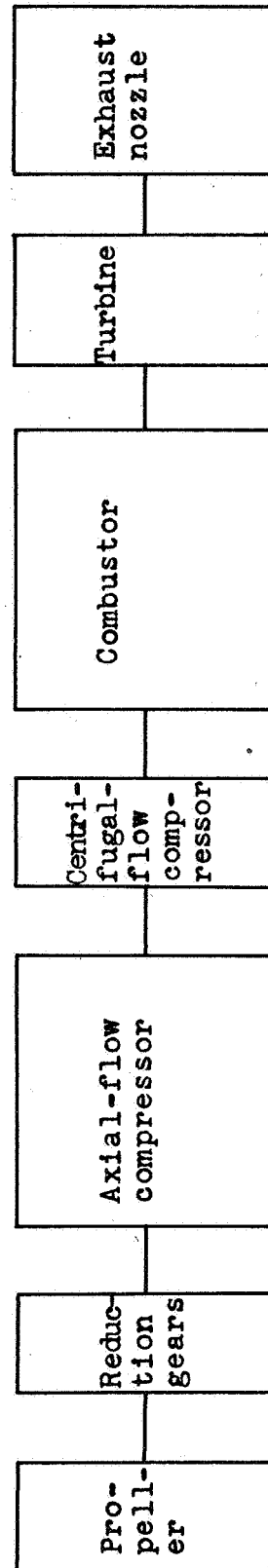
D_n	nacelle drag per unit engine frontal area, lb/sq ft
F	net thrust per unit engine frontal area, lb/sq ft
f	thrust specific fuel consumption, lb/(lb)(hr)
K	ratio of average fuel rate to initial fuel rate
L/D	lift-drag ratio of aircraft without nacelles
P	total pressure at compressor inlet, in. Hg
P_o	sea-level zero-ram pressure, in. Hg
R	range, miles
r_c	compressor pressure ratio
T	total temperature at compressor inlet, °R
T_o	sea-level zero-ram temperature, °R
V	flight speed, mph
W_c	pay load per unit engine frontal area, lb/sq ft
W_d	disposable load per unit engine frontal area, lb/sq ft
W_e	engine weight per unit engine frontal area, lb/sq ft
W_f	fuel plus fuel-tank weight per unit engine frontal area, lb/sq ft
W_f'	initial fuel rate per unit engine frontal area, lb/(mi)(sq ft)
W_g	gross weight per unit engine frontal area, lb/sq ft
W_s	structure weight per unit engine frontal area, lb/sq ft
w	air flow per unit engine frontal area, lb/sec
η_p	propeller efficiency

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2. Lubarsky, Bernard: Performance and Load-Range Characteristics of Turbojet Engine in Transonic Speed Range. NACA TN 2088, 1950.
3. Keenan, Joseph H., and Kaye, Joseph: Gas Tables. John Wiley & Sons, Inc., 1948.
4. Pinkel, Benjamin, and Karp, Irving M.: A Thermodynamic Study of the Turbojet Engine. NACA Rep. 891, 1947.
5. Pinkel, Benjamin, and Turner, L. Richard: Thermodynamic Data for the Computation of the Performance of Exhaust-Gas Turbines. NACA ARR 4B25, 1944.
6. Trout, Arthur M., and Hall, Eldon W.: Method for Determining Optimum Division of Power between Jet and Propeller for Maximum Thrust Power of a Turbine-Propeller Engine. NACA TN 2178, 1950



(a) Compressor pressure ratio less than or equal to 10.



(b) Compressor pressure ratio greater than 10.

Figure 1. - Diagram of assumed turbine-propeller engine.



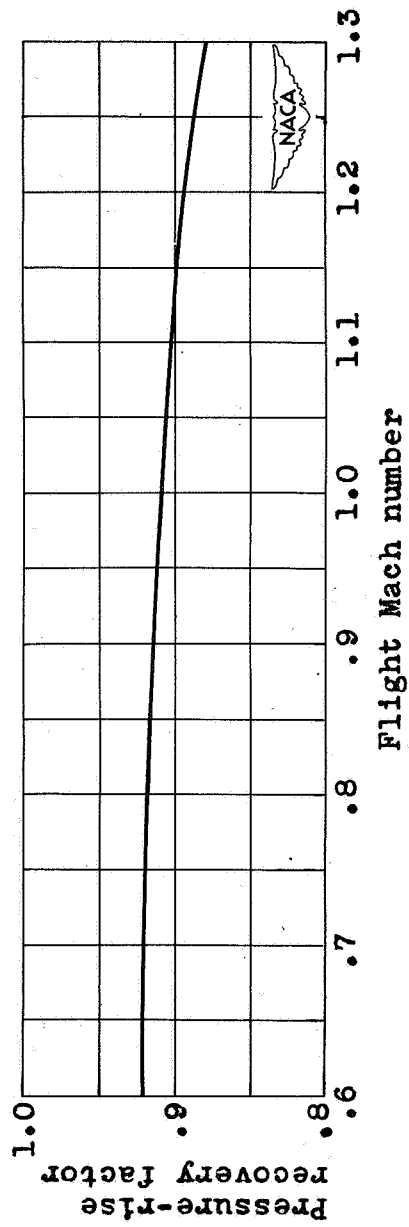


Figure 2. - Variation of recovery factor of engine-inlet-diffuser pressure rise with flight Mach number.

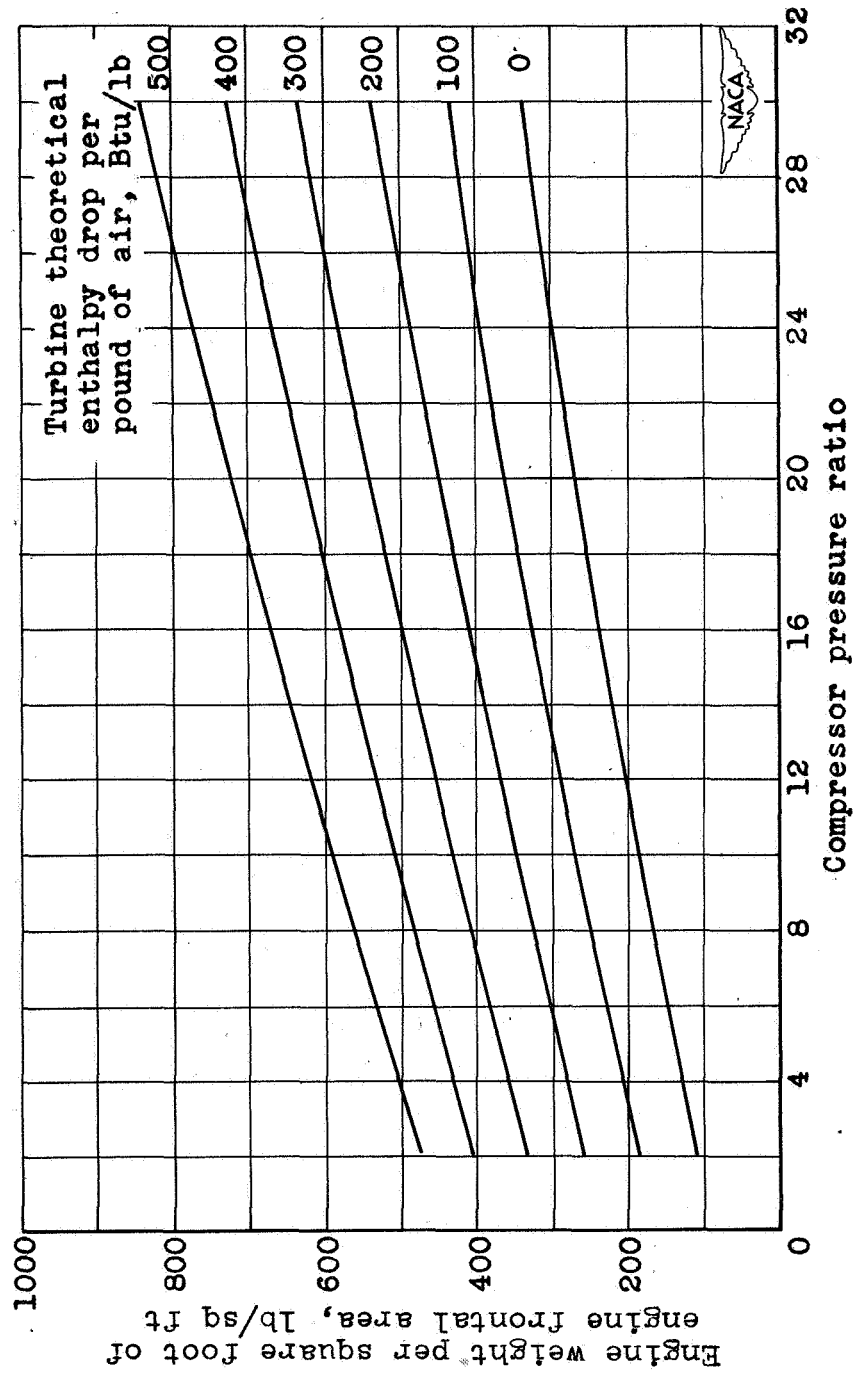


Figure 3. - Variation of engine weight (without propeller and gears) with compressor pressure ratio and turbine enthalpy drop.

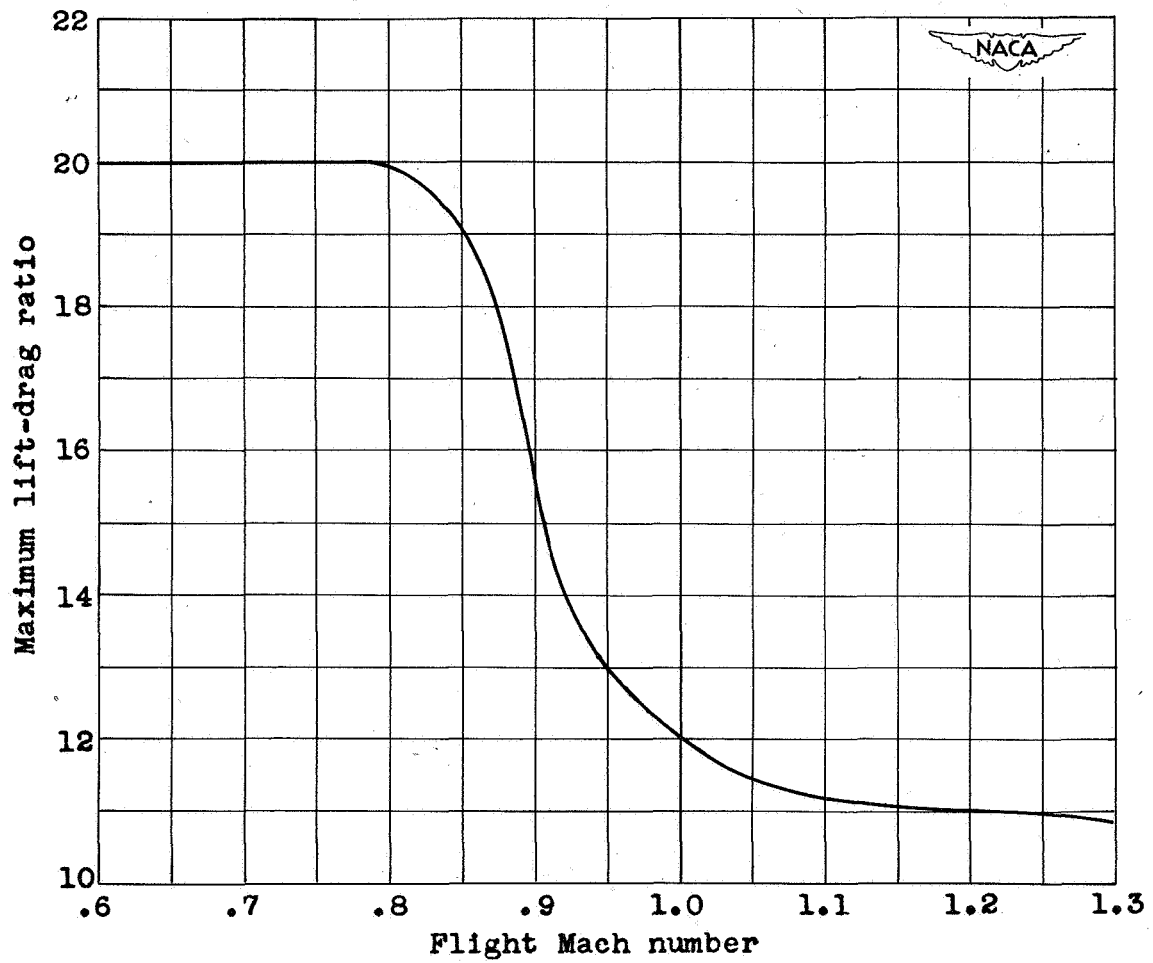


Figure 4. - Variation of maximum airplane lift-drag ratio with flight Mach number.

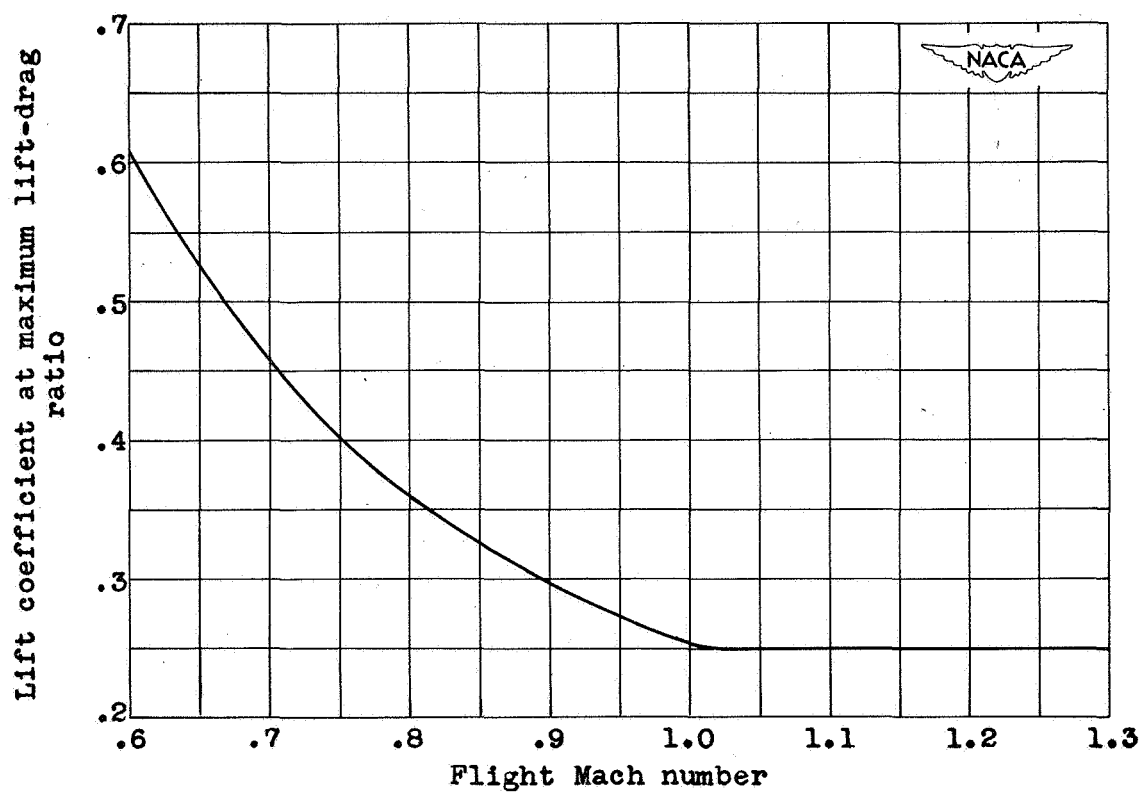


Figure 5. - Variation of lift coefficient at maximum airplane lift-drag ratio with flight Mach number.

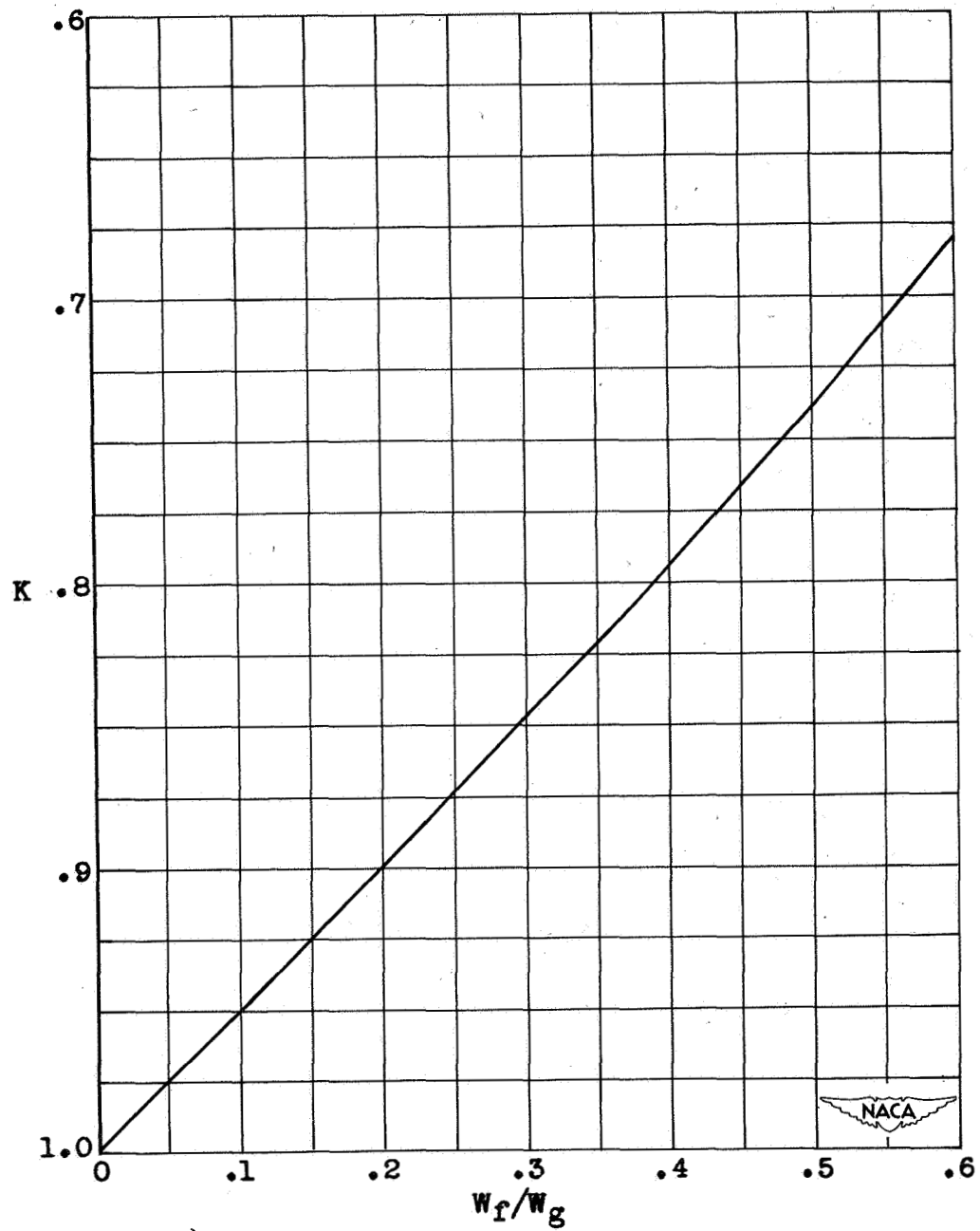
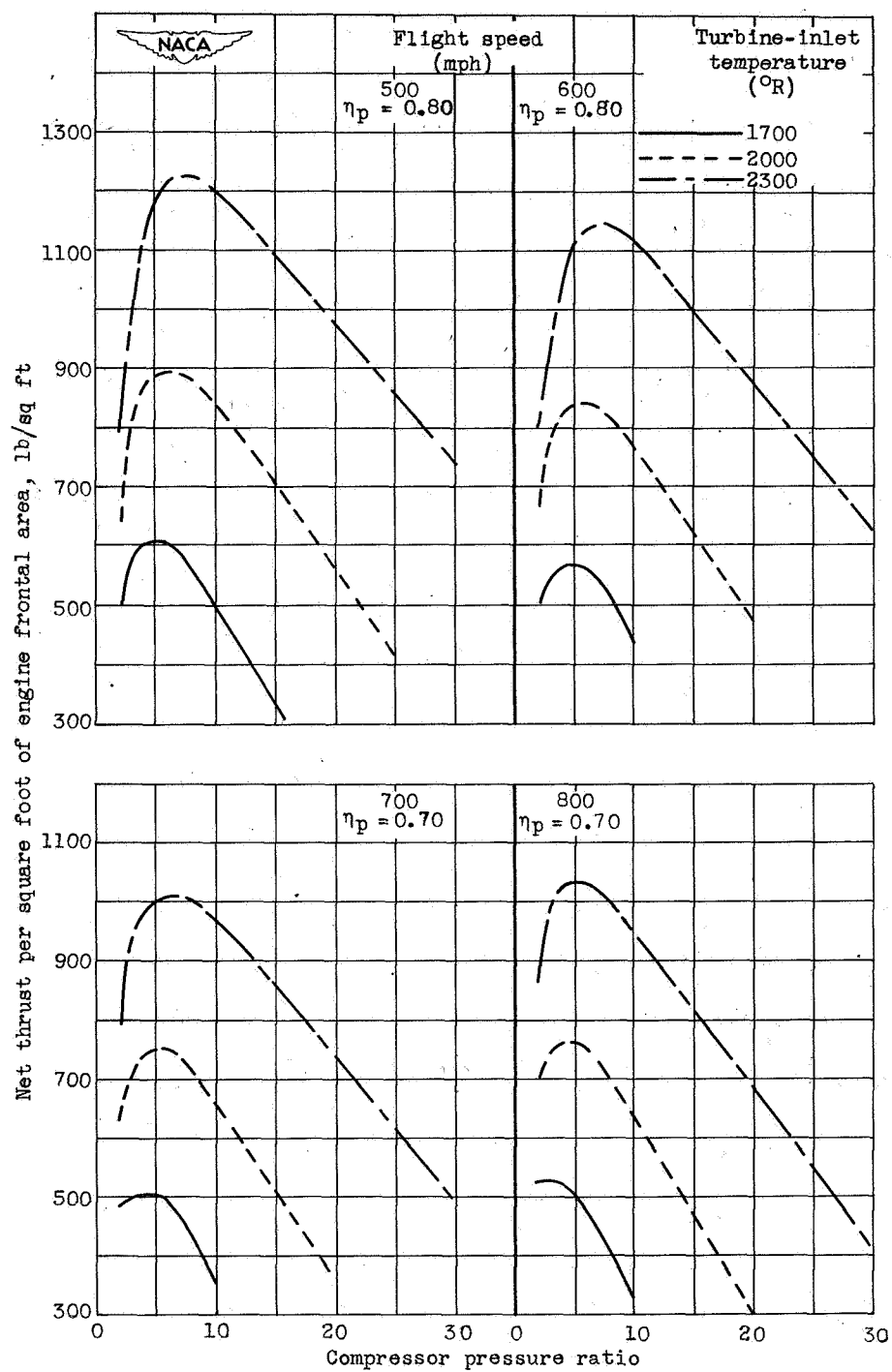


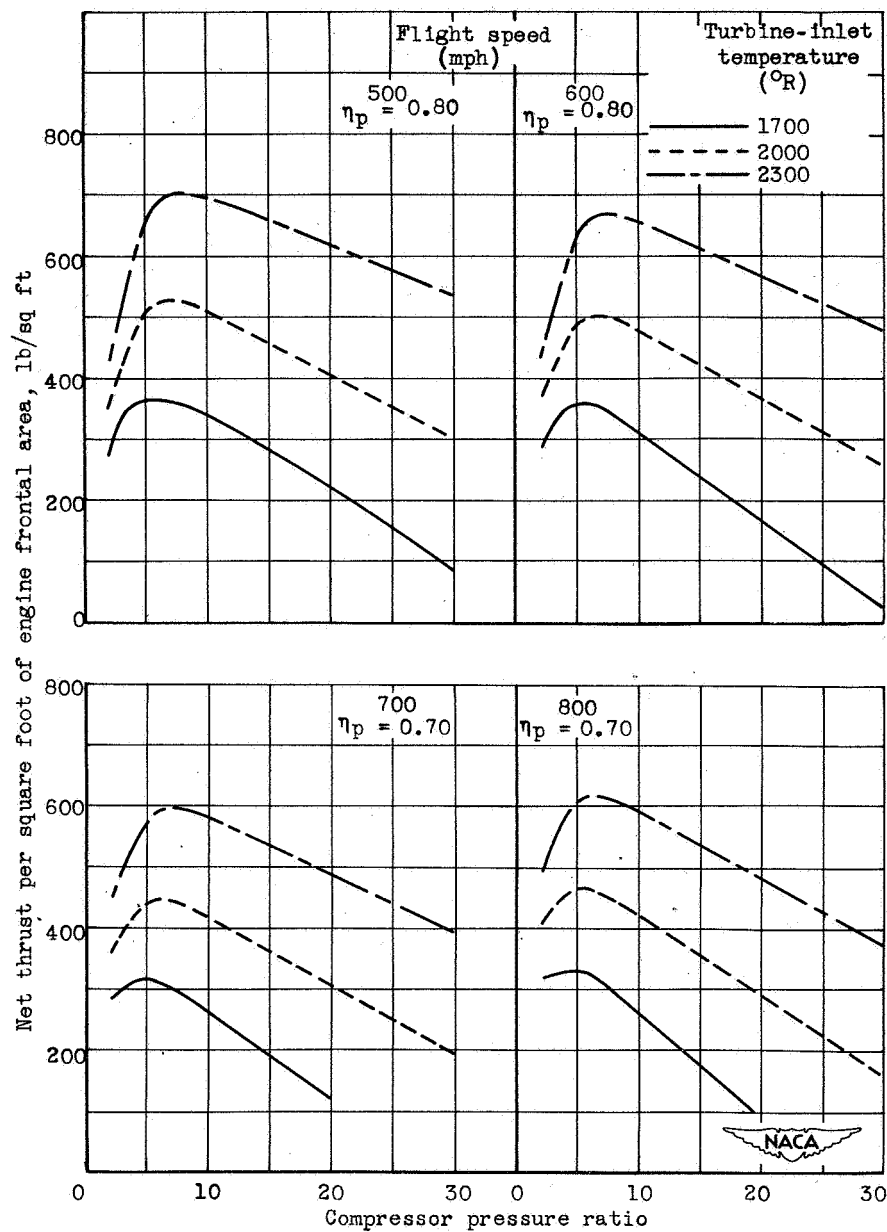
Figure 6. - Variation of K with W_f/W_g .

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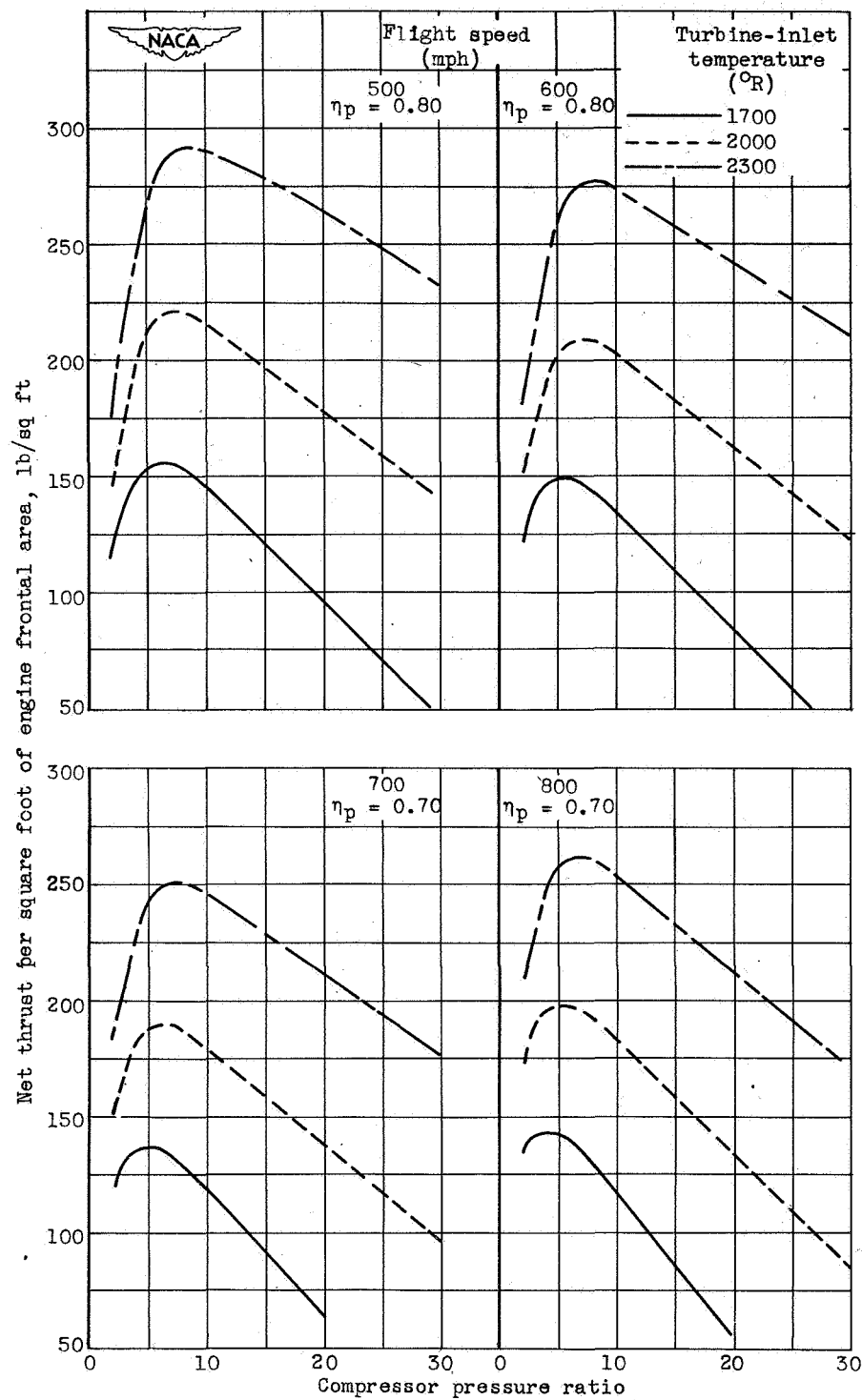
(a) Altitude, 10,000 feet.

Figure 7. - Variation of net thrust per square foot of engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



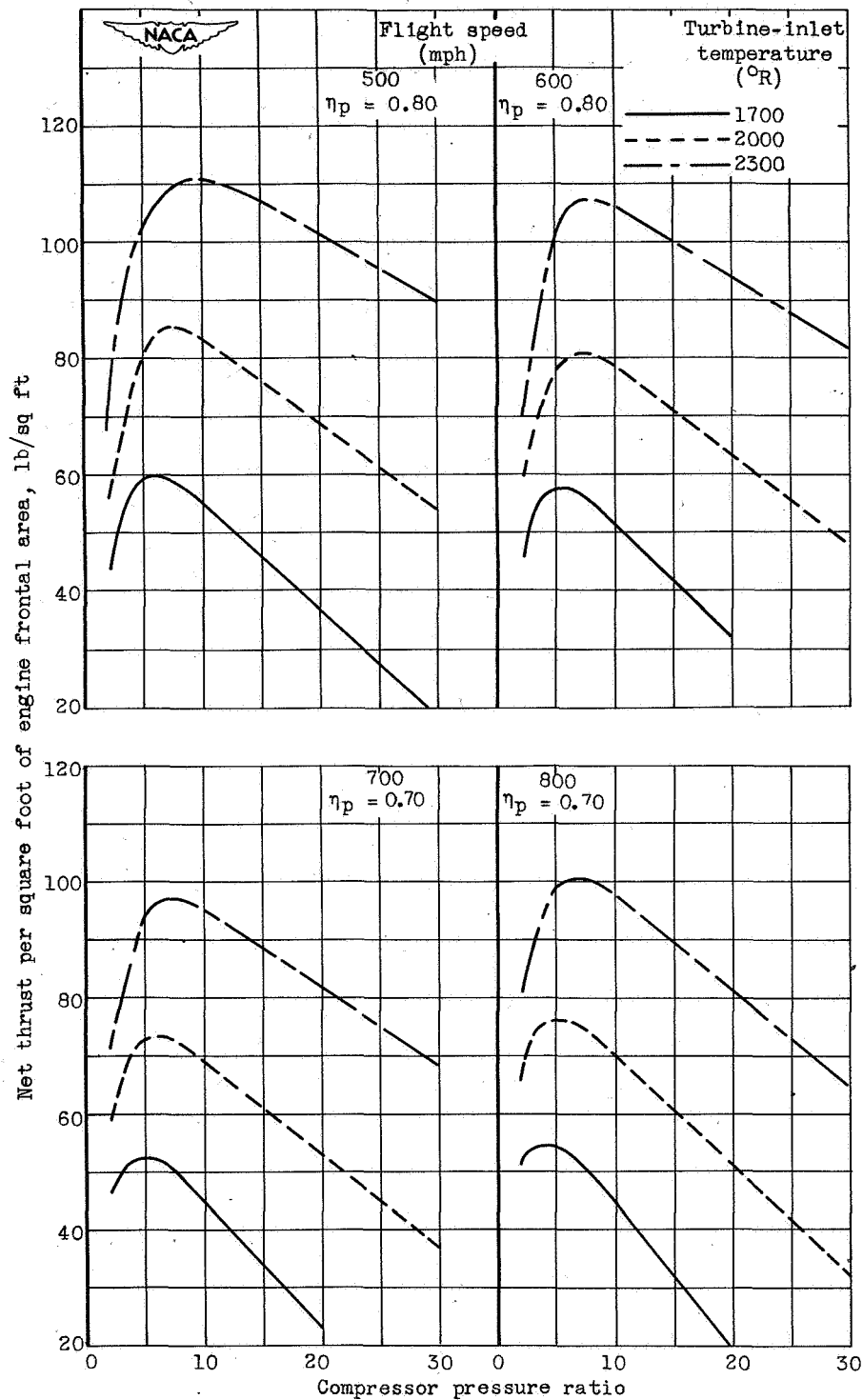
(b) Altitude, 30,000 feet.

Figure 7. - Continued. Variation of net thrust per square foot of engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



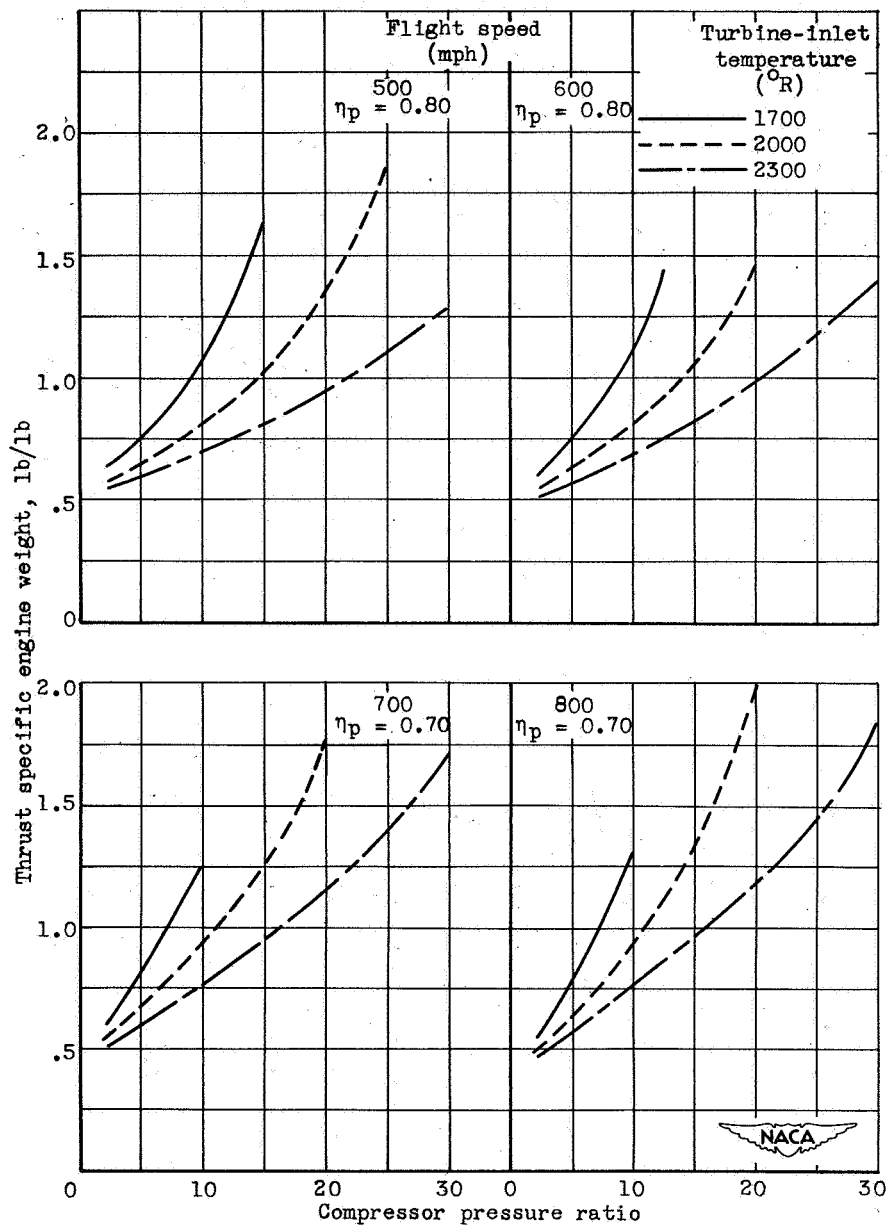
(c) Altitude, 50,000 feet.

Figure 7. - Continued. Variation of net thrust per square foot of engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



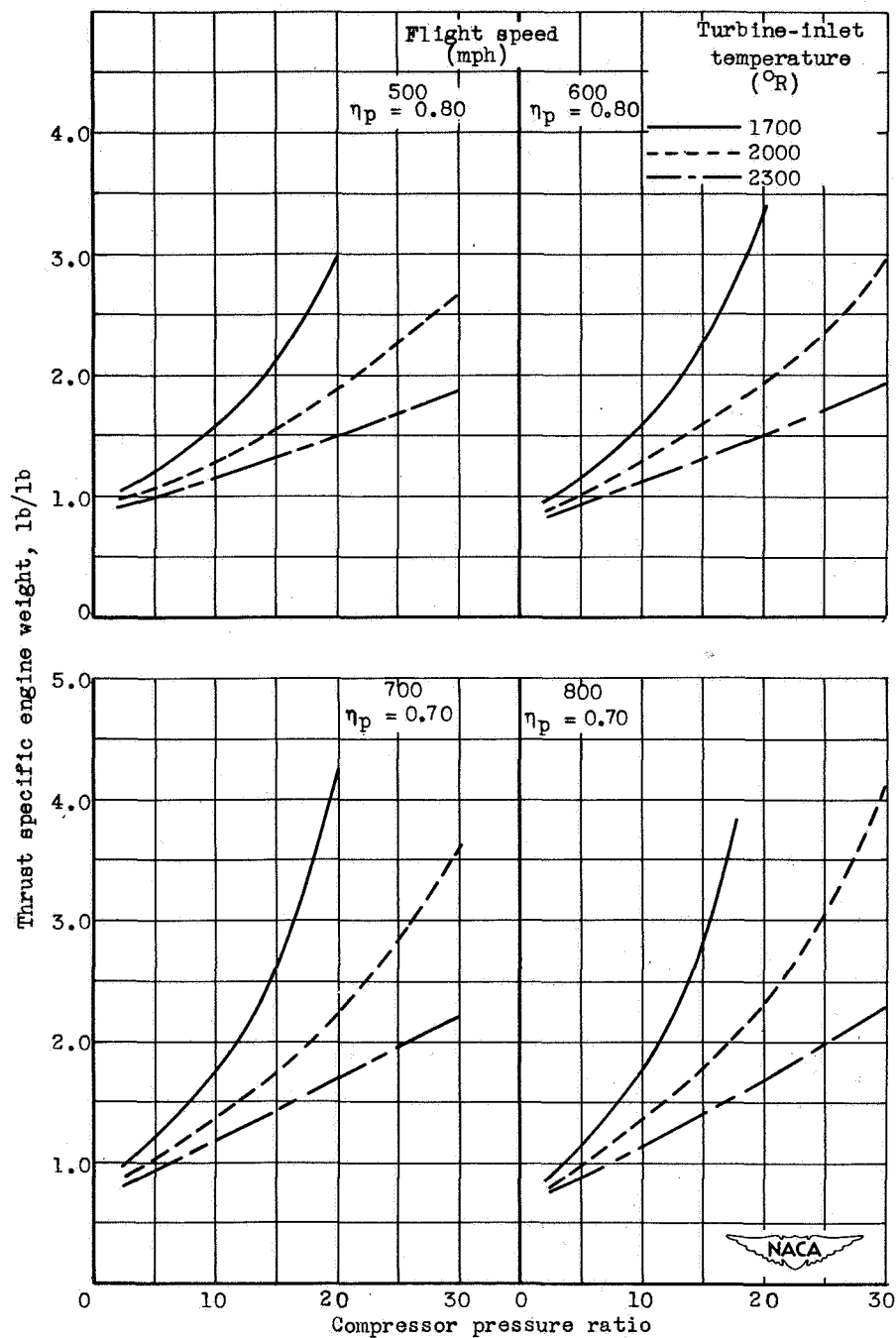
(d) Altitude, 70,000 feet.

Figure 7. - Concluded. Variation of net thrust per square foot of engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



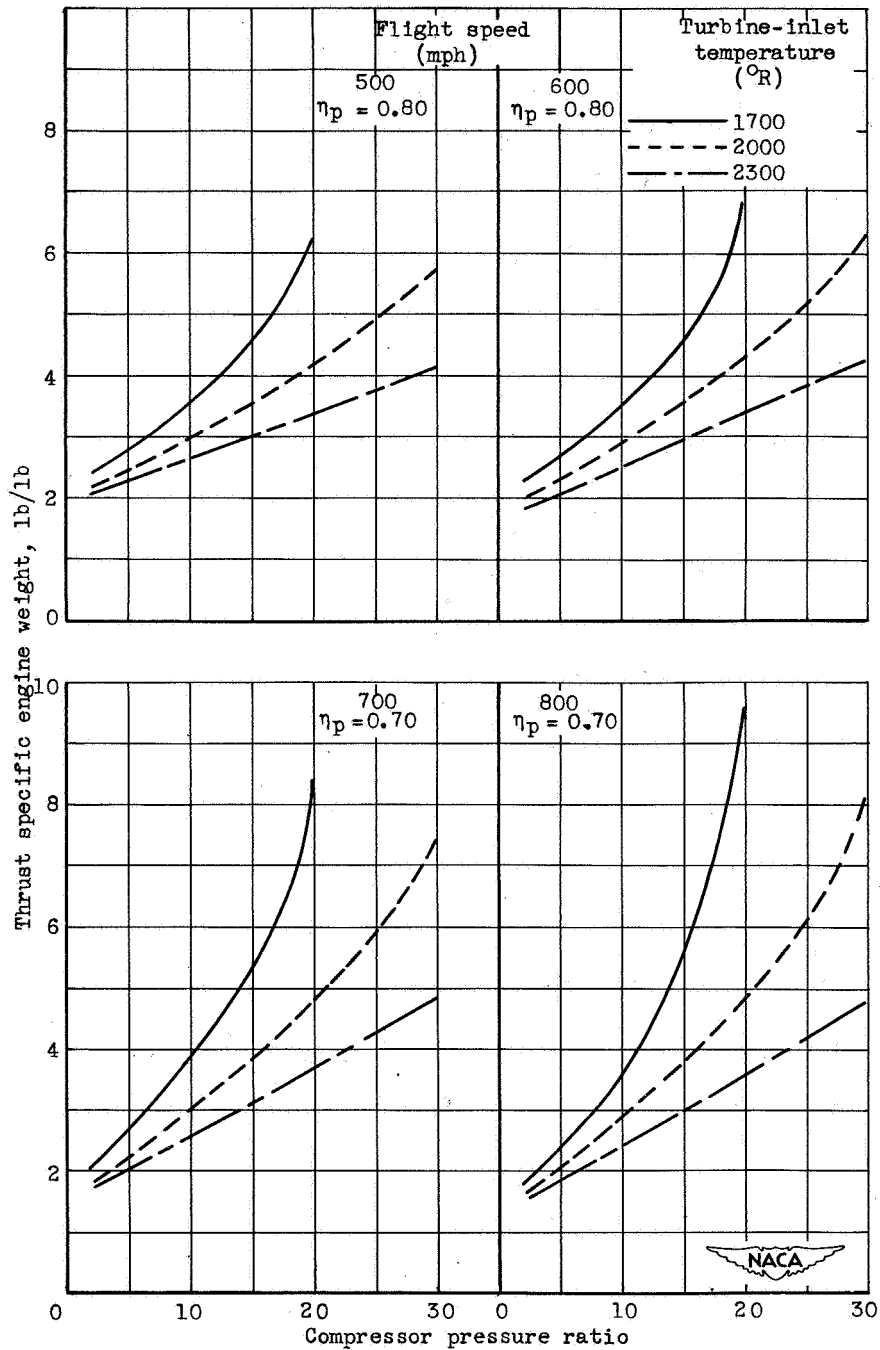
(a) Altitude, 10,000 feet.

Figure 8. - Variation of specific engine weight (lb engine per lb thrust) with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



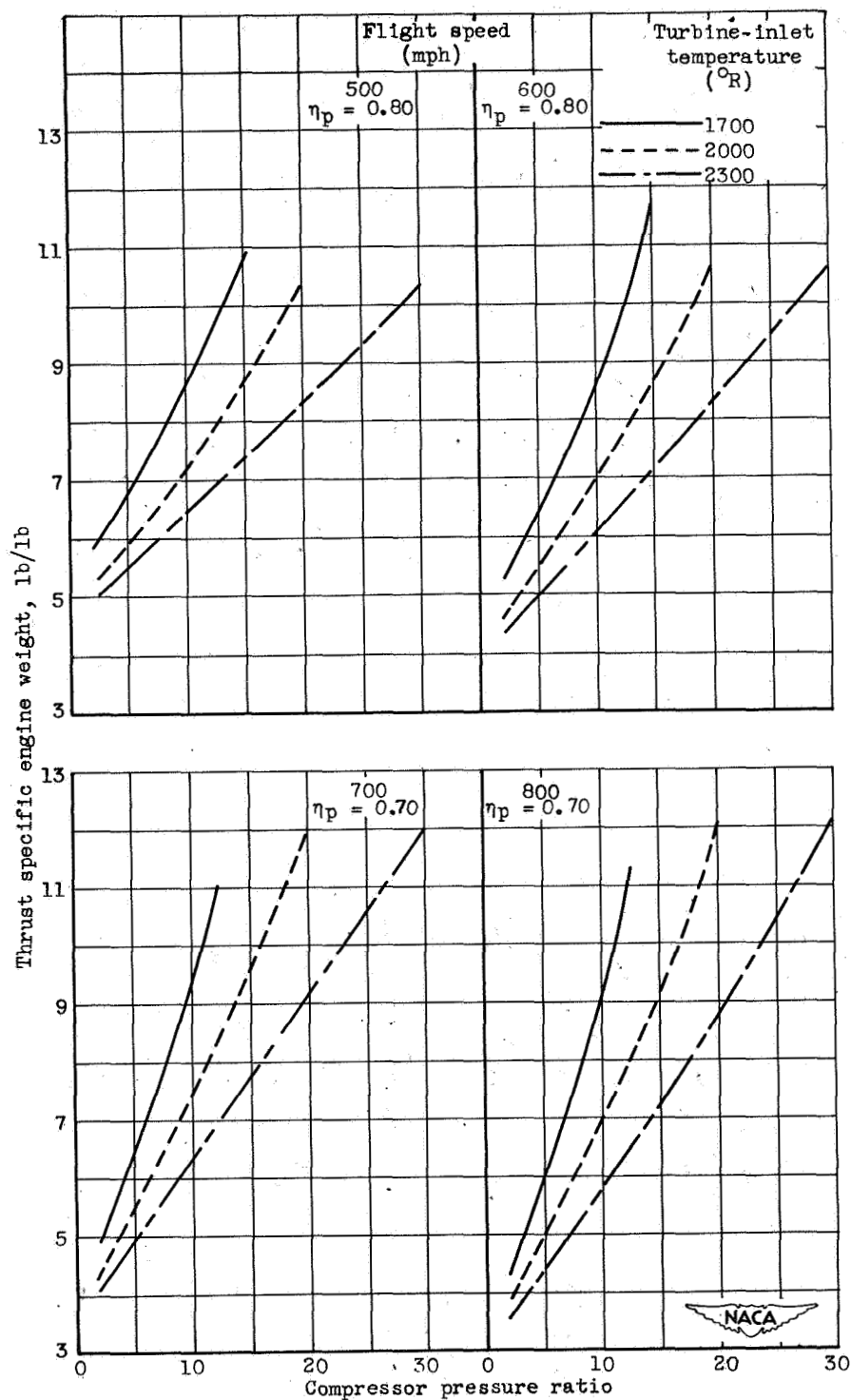
(b) Altitude, 30,000 feet.

Figure 8. - Continued. Variation of specific engine weight (lb engine per lb thrust) with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



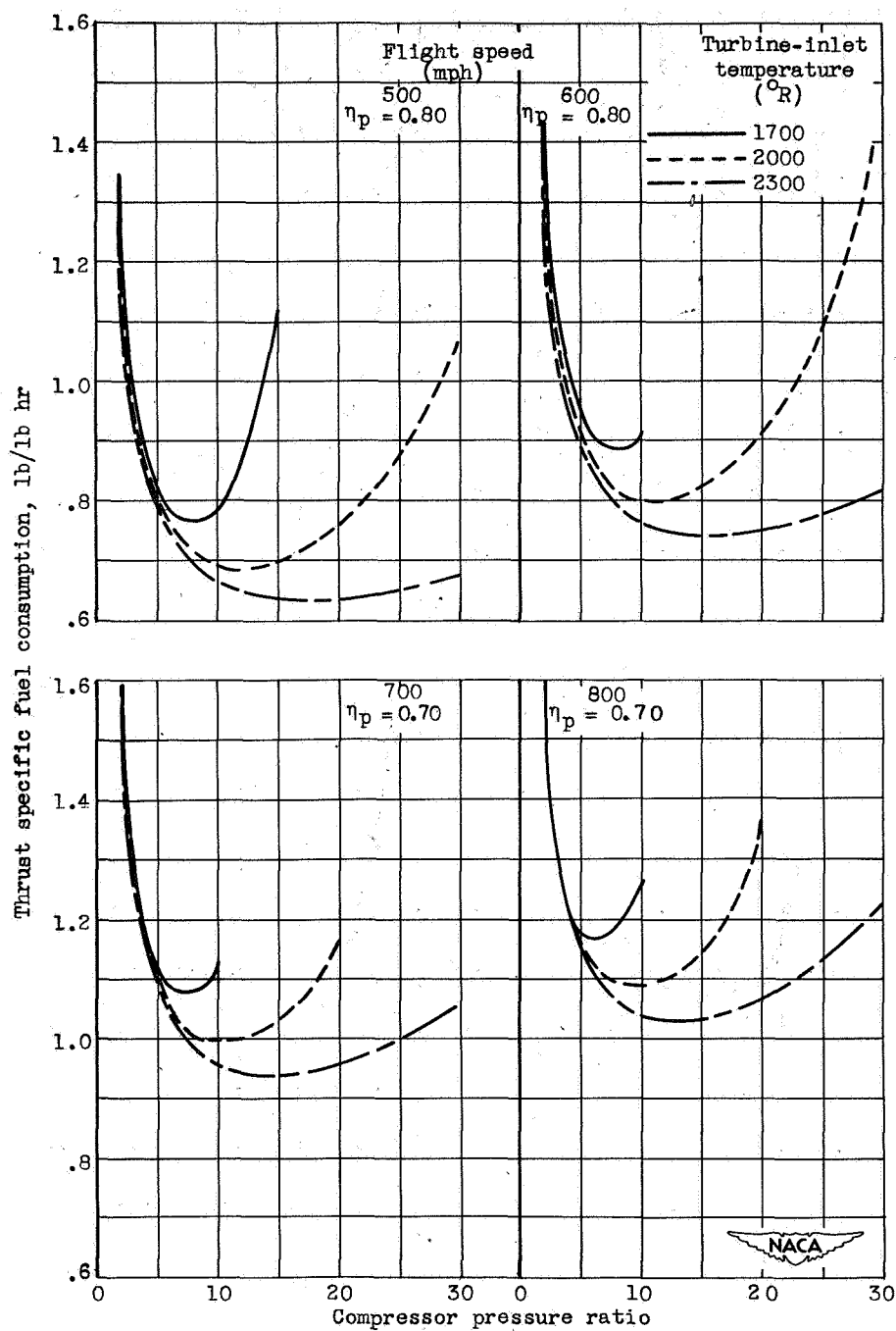
(c) Altitude, 50,000 feet.

Figure 8. - Continued. Variation of specific engine weight (lb engine per lb thrust) with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



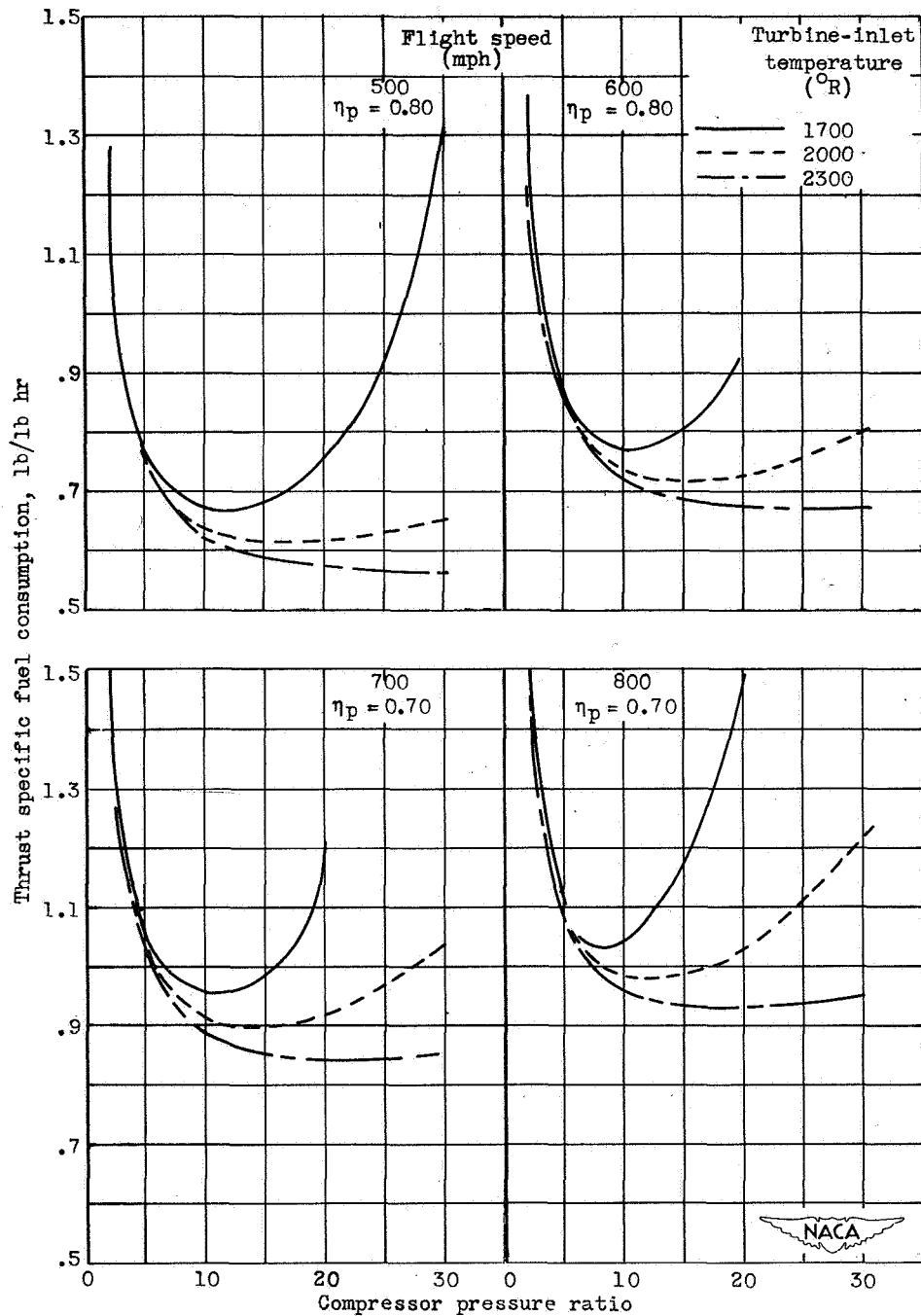
(d) Altitude, 70,000 feet.

Figure 8. - Concluded. Variation of specific engine weight (lb engine per lb thrust) with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



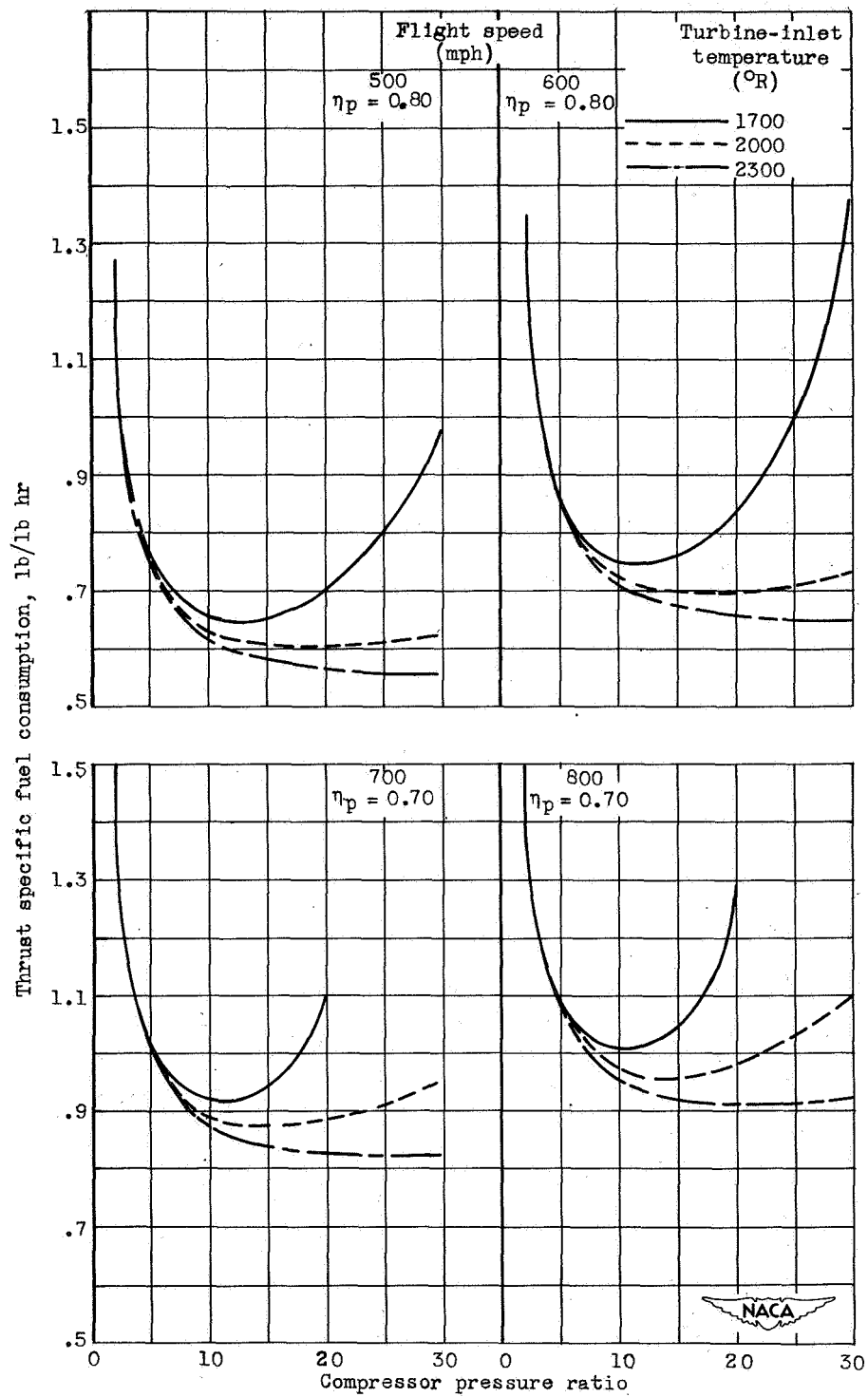
(a) Altitude, 10,000 feet.

Figure 9. - Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(b) Altitude, 30,000 feet.

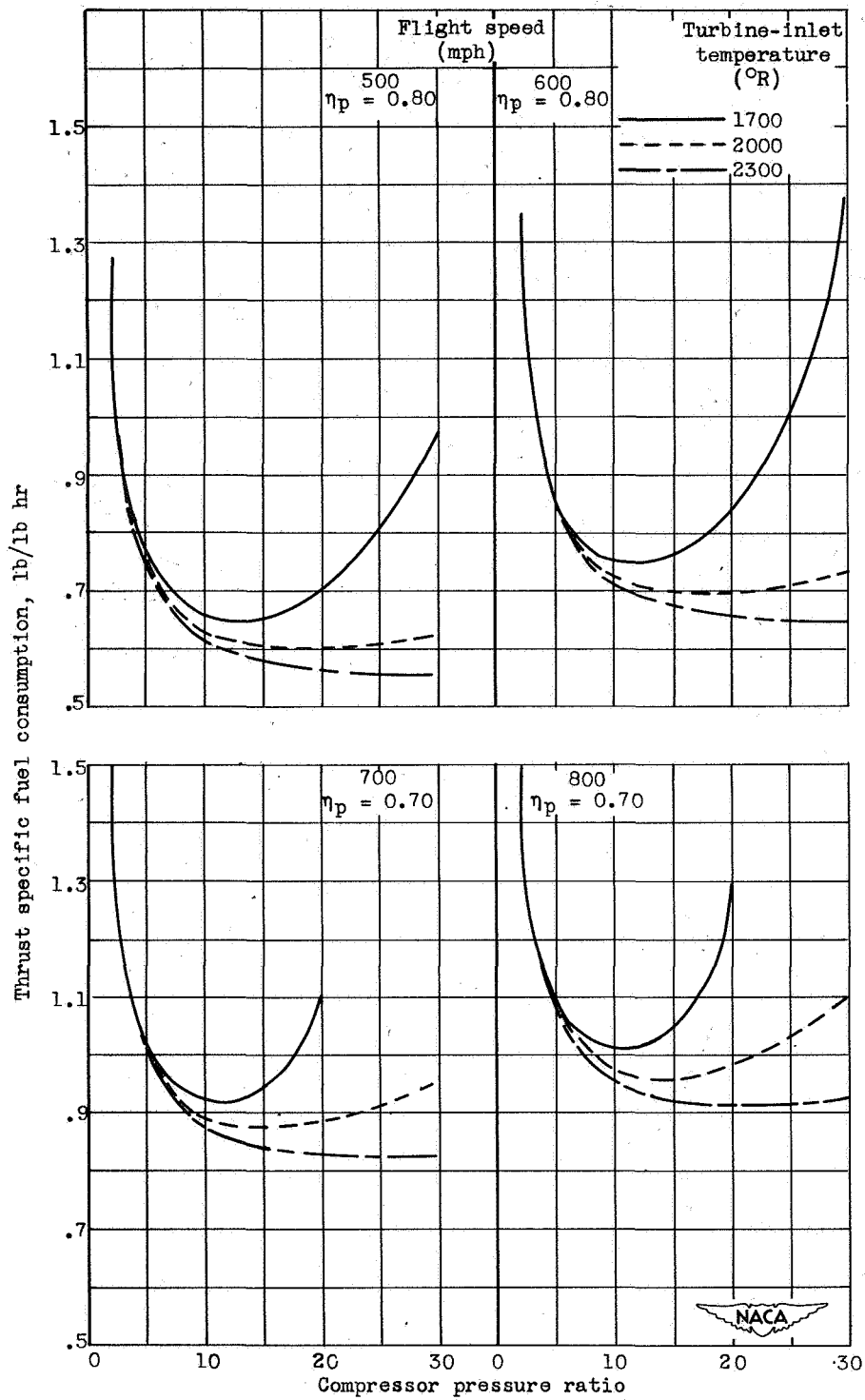
Figure 9. - Continued. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(c) Altitude, 50,000 feet.

Figure 9. - Continued. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

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(d) Altitude, 70,000 feet.

Figure 9. - Concluded. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

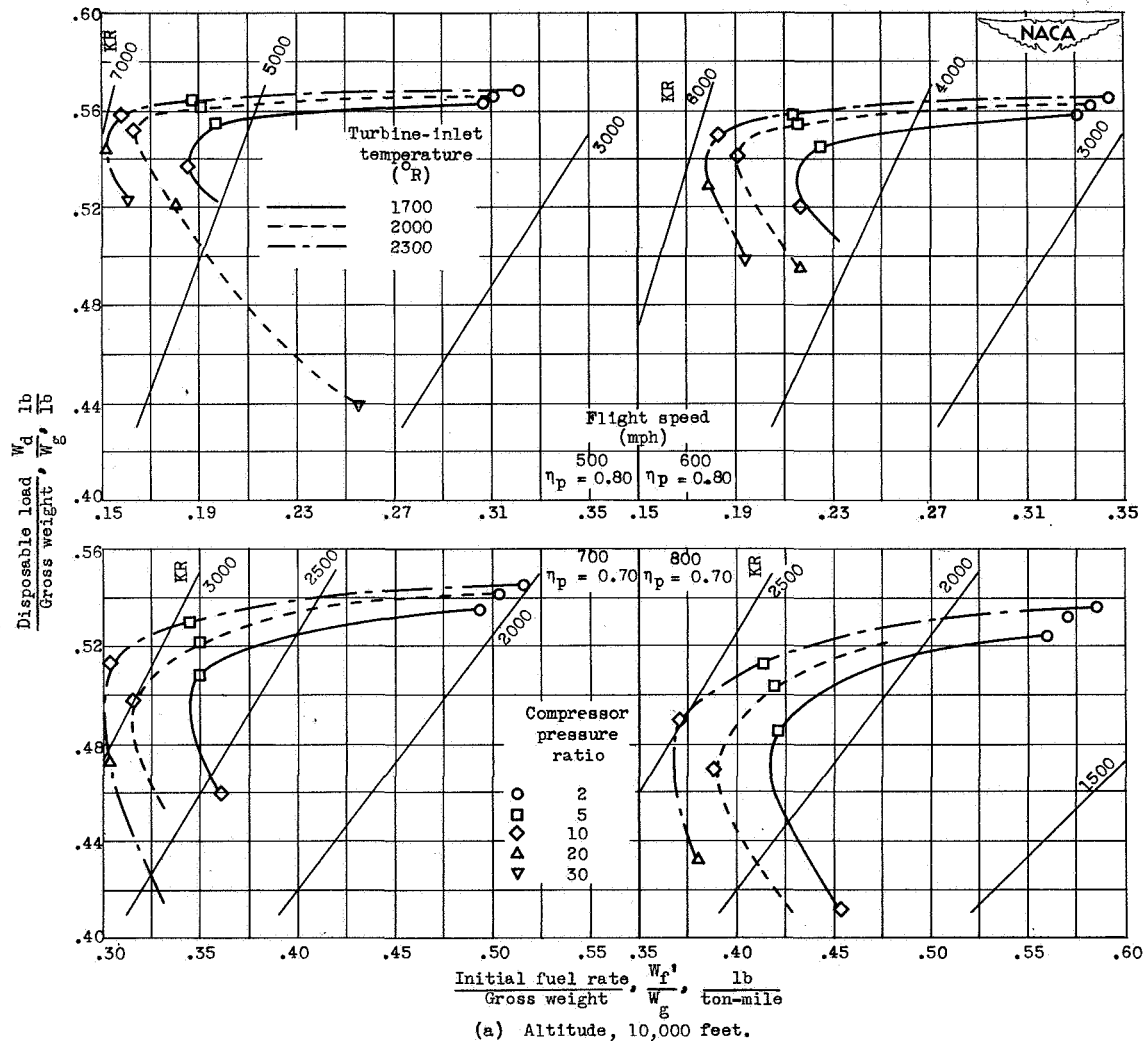


Figure 10. - Load-range characteristics of turbine-propeller engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

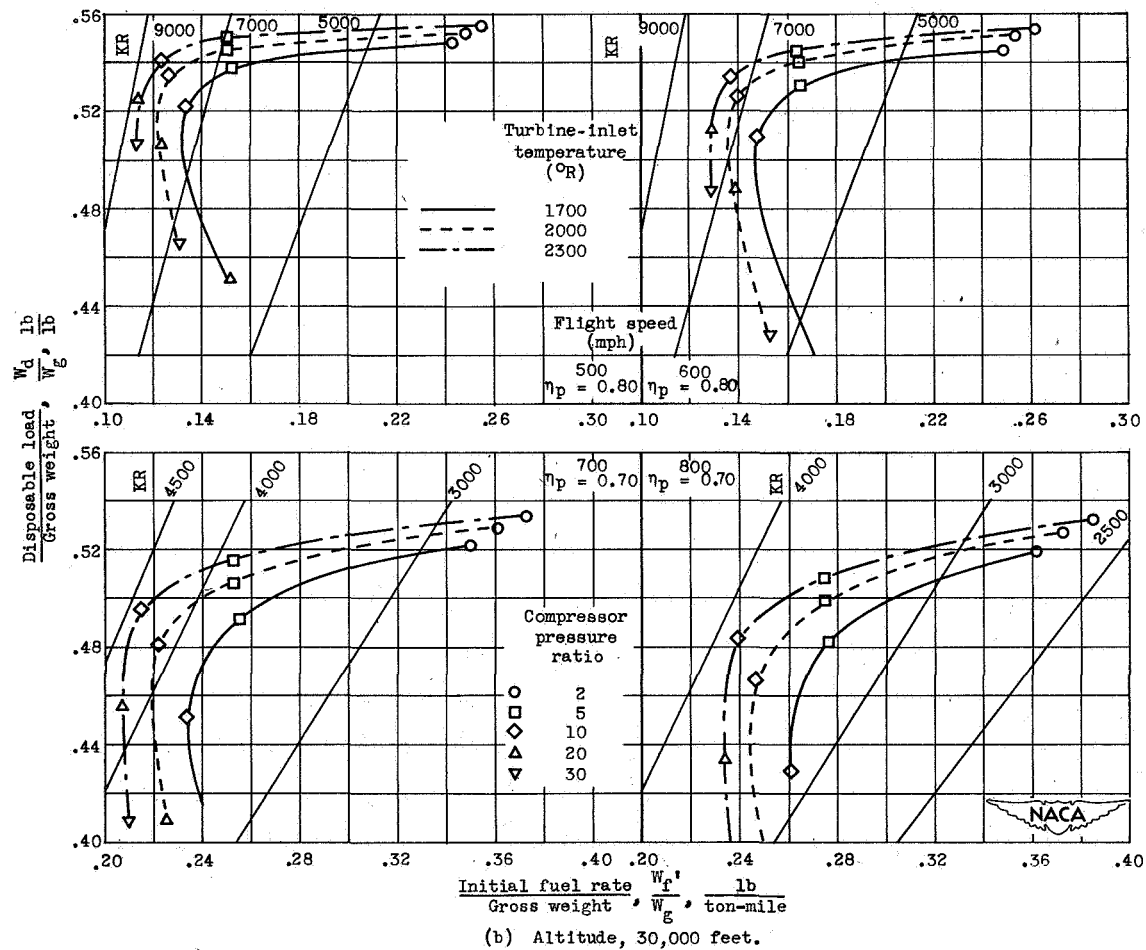


Figure 10. - Continued. Load-range characteristics of turbine-propeller engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

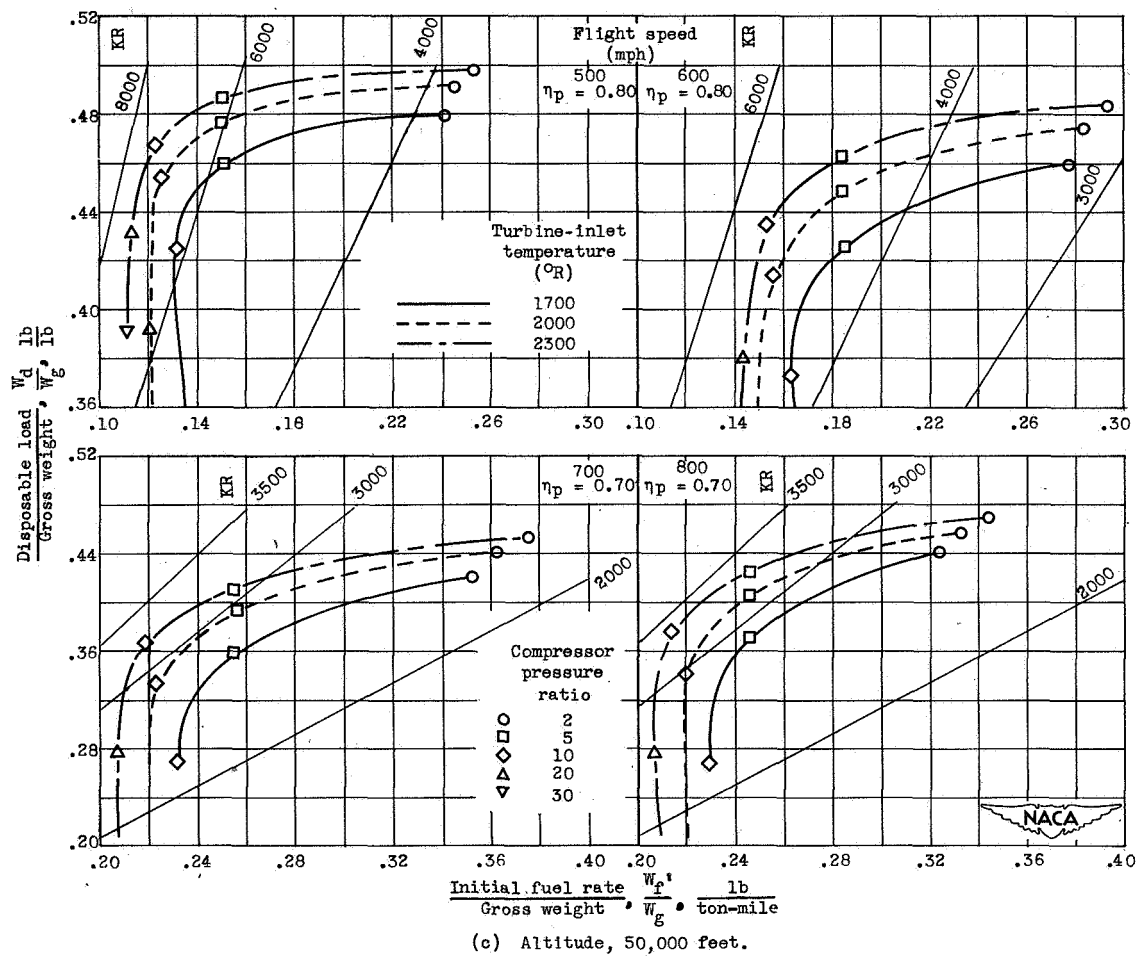


Figure 10. - Continued. Load-range characteristics of turbine-propeller engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

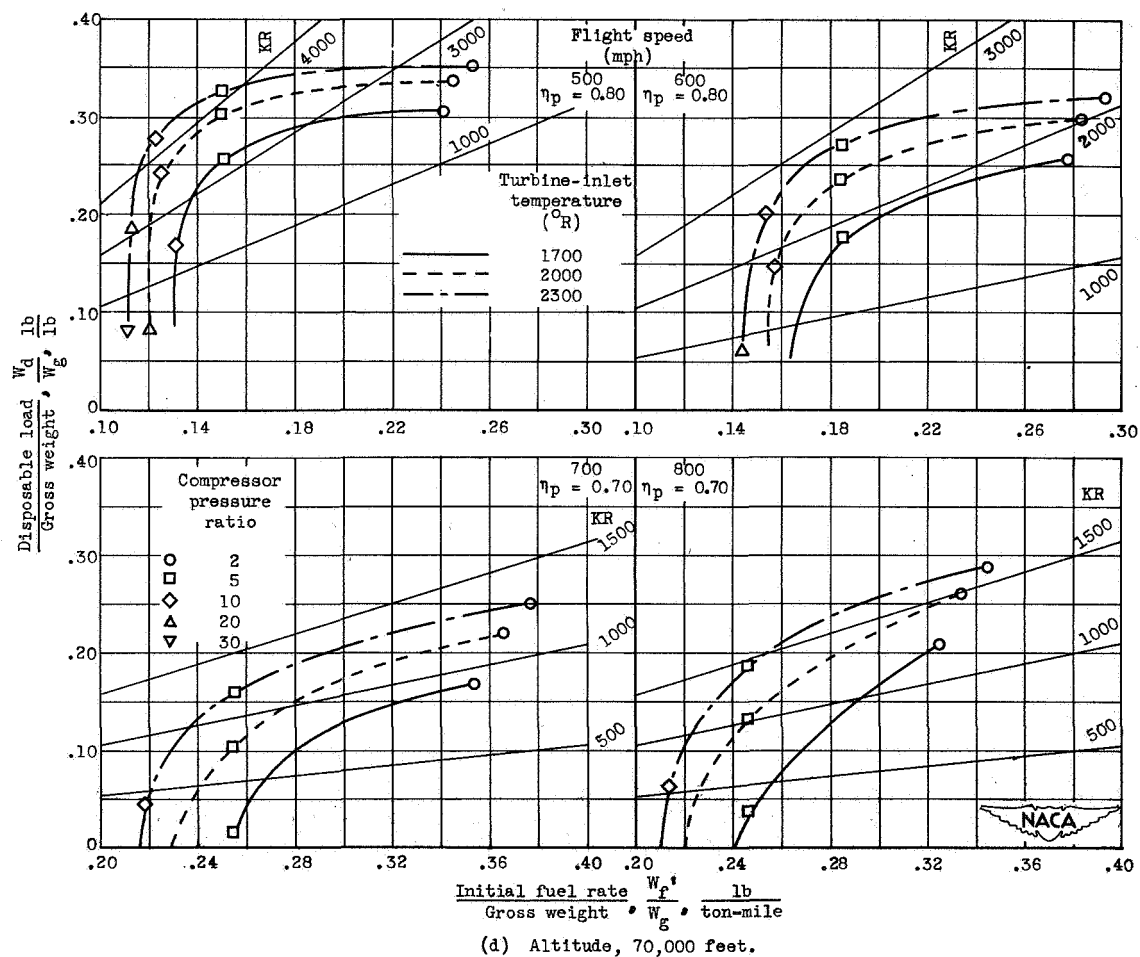


Figure 10. - Concluded. Load-range characteristics of turbine-propeller engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

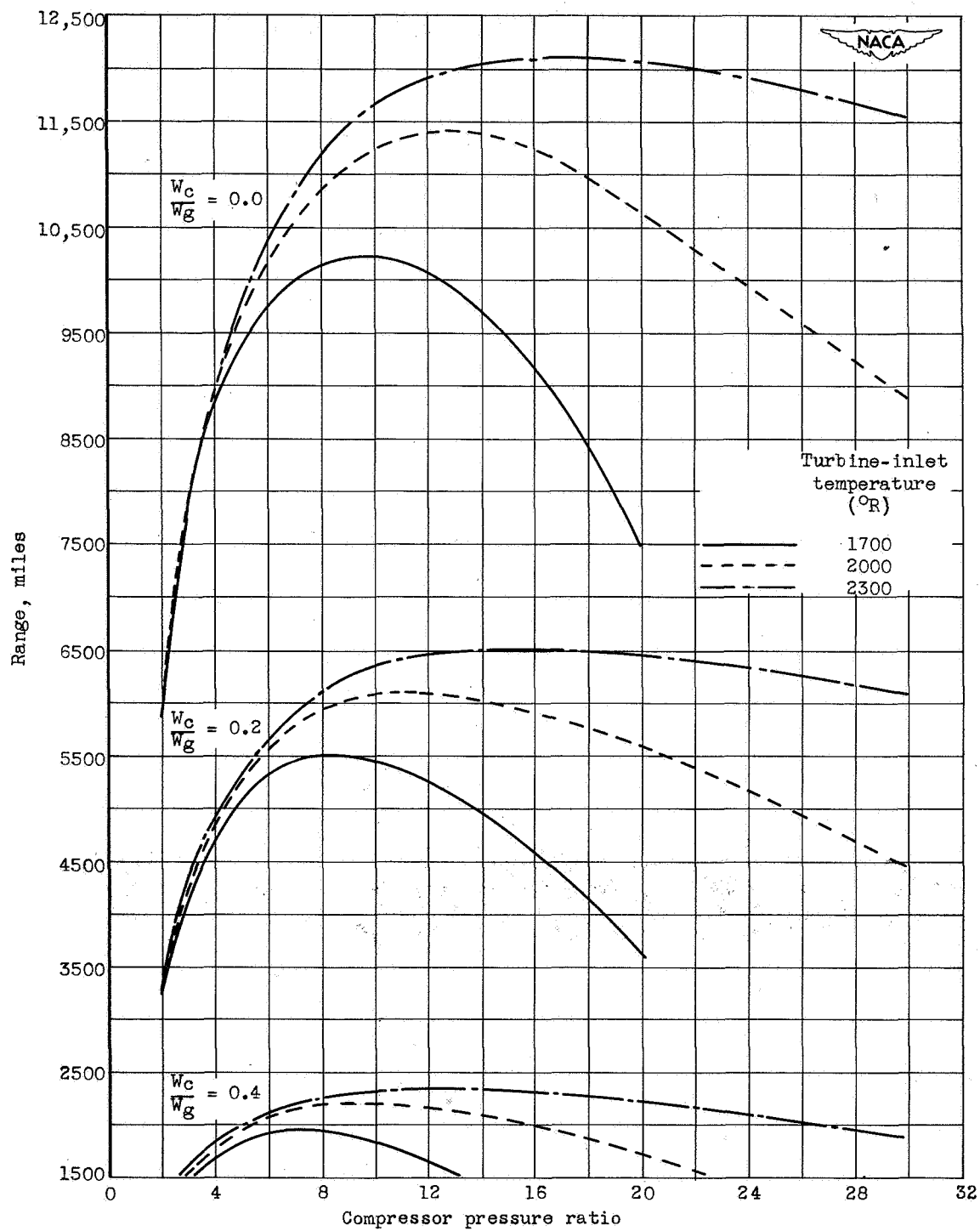


Figure 11. - Variation of range with compressor pressure ratio at 30,000 feet and 500 miles per hour for various pay loads and turbine-inlet temperatures.

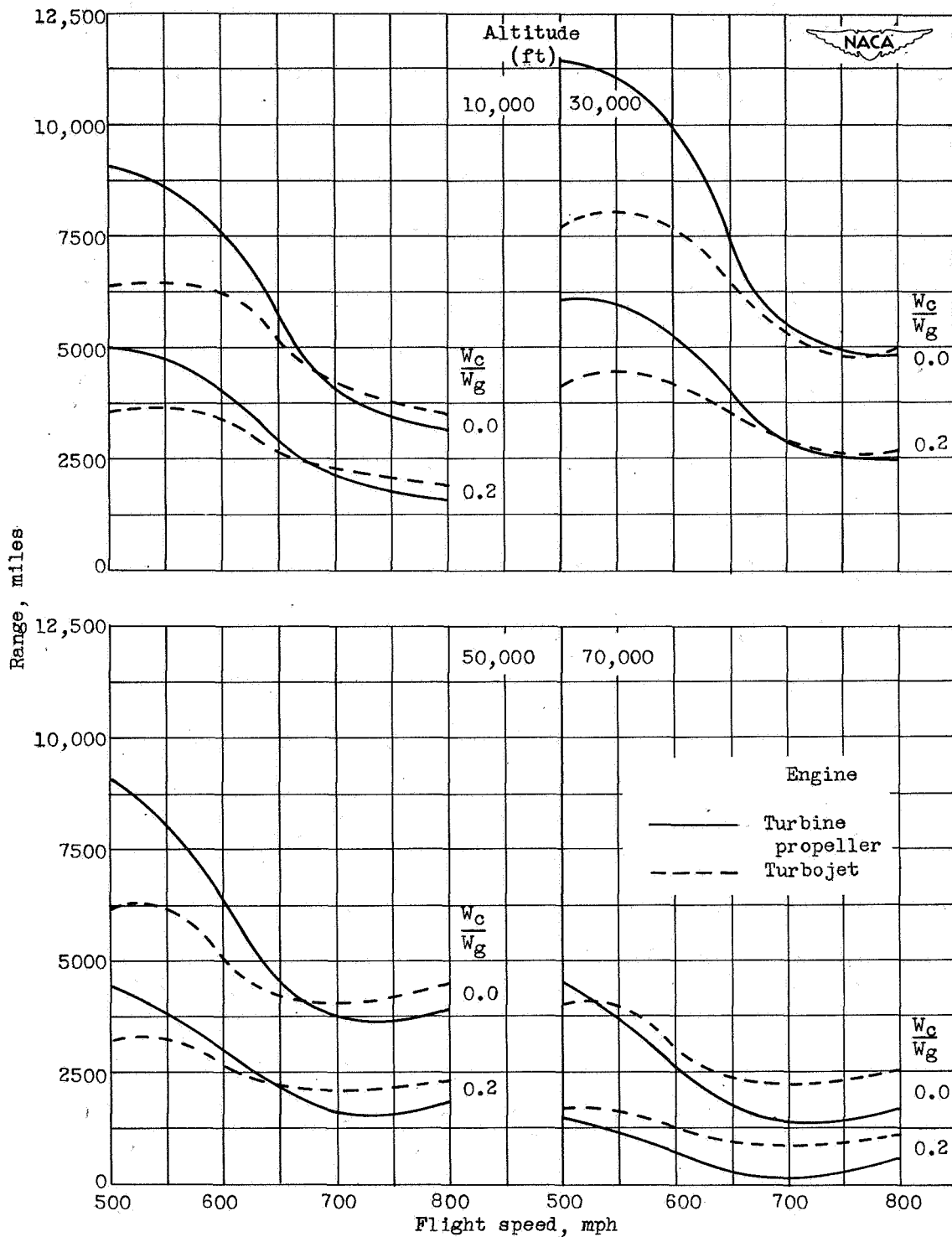


Figure 12. - Comparison of variation of range of turbine-propeller and turbojet engines with flight speed at various altitudes and pay loads with turbine-inlet temperature of 2000°R and compressor pressure ratio giving longest range in each particular case.

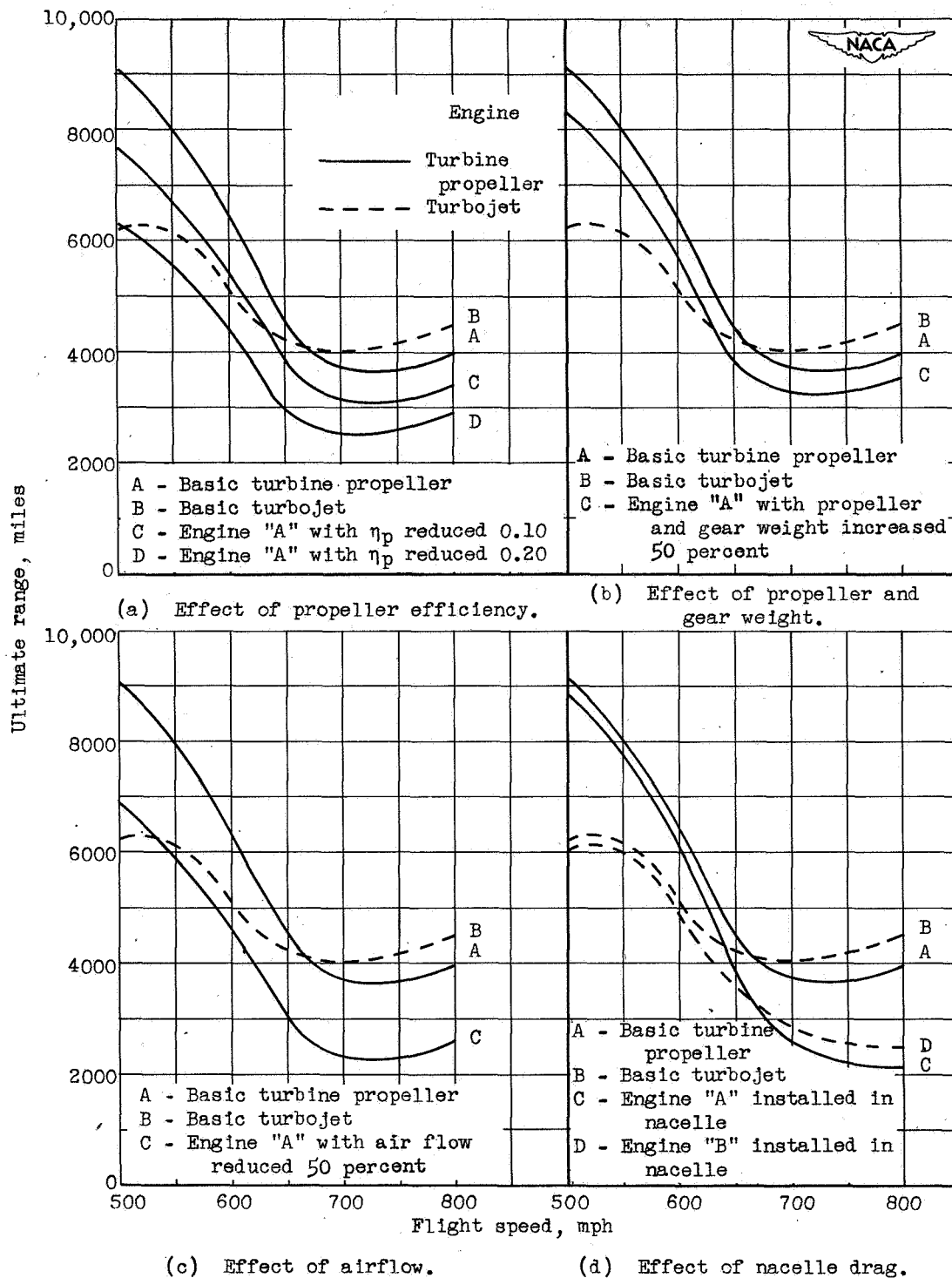


Figure 13. - Effect of changes in some assumptions on comparison of variation with flight speed of ultimate range of turbine-propeller and turbojet engines at altitude of 50,000 feet, with turbine-inlet temperature of 2000° R and compressor pressure ratio giving longest range in each particular case.